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LAPR 64-71

INVESTIGATION OF ENGINE-COMPONENT INTEGRATION

VOLUME II
FINAL REPORT

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ROCKETDYNE
A DIVISION OF NORTH AMERICAN AVIATION, INC.

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INVESTIGATION OF ENGINE-COMPONENT INTEGRATION

VOLUME II
FINAL REPORT

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ROCKETDYNE

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First Assistant Commissioner

FOREWORD

This report was prepared in compliance with the requirements for the National Aeronautics and Space Administration Contract, NAS 8-4001, Investigation of Engine-Component Integration Study. The NASA technical monitors have been Mr. J. McCarty at the Marshal Space Flight Center, and Mr. J. Suddreth at the NASA headquarters.

ABSTRACT

(Unclassified Abstract)

The analyses and results of the investigations conducted under the National Aeronautics and Space Administration Contract, NAS 8-4001, Investigation of Engine-Component Integration Study, are presented in this report. Component concepts for functional and packaging integration were investigated for spacecraft and boosters. Spacecraft propellant combinations considered were O_2/H_2, F_2/H_2, and N_2O_4/N_2H_4-UDMH(50-50); booster propellants were O_2/H_2 and O_2/RP-1. Evaluations were made of a number of system concepts, with and without turbopumps. Preliminary-design layouts were made of the more-promising concepts.

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INTRODUCTION

The superior simplicity and reliability of pressure-fed systems are frequently the reasons for their selection in applications which would otherwise suggest the use of pump-fed systems. Examples of such applications are those wherein, small propulsion-system volumes and/or long firing durations are desirable (or required), but the additional requirements of maximum simplicity and reliability preclude the use of typical pump-fed systems.

This study was initiated to investigate methods of making pump-fed systems more competitive with the simplicity of pressure-fed systems, and to investigate other concepts which could possess the desirable characteristics of both pump-fed and pressure-fed systems, but would not necessarily use conventional methods of pumping.

More specifically the objective was to evolve systems which, because of their more efficient use of volume; possess greater operational-simplicity, lower weights, and higher reliabilities than can be attained by conventional pump-fed systems.

Concepts to be investigated were to be applicable to advanced booster systems using LO_2/LH_2 and LO_2/RP-1; and to spacecraft systems using LO_2/LH_2, LF_2/LH_2 and N_2O_4/N_2H_4-UDMH(50-50) as propellants. Spacecraft systems were to be considered for both conventional and advanced nozzles.

The more-promising system and/or component concepts were to be selected for preliminary design and layout.

SUMMARY

Functional system configurations were defined and rated on the basis of operational simplicity, weight and reliability. Further investigations and preliminary-design layouts were made for the optimum systems selected on the basis of the above criteria. These investigations were of two types, namely, analytical and design.

The analytical investigations were primarily directed at defining and evaluating (1) novel methods of functionally integrating somewhat conventional components, and (2) novel methods of accomplishing the same function as the basic systems, but with fewer and/or less complex components. This latter effort consisted largely of evaluating methods of obtaining a high-chamber-pressure capability without using turbopumps or increasing propellant-tank pressure.

The design investigations were primarily directed at defining novel methods of physically integrating the components of the basic systems. This generally produced some feedback to the basic systems, and the basic systems were modified as these investigations indicated how further functional simplification could be achieved.

The effort to functionally integrate somewhat conventional components has resulted in significant system simplification. This simplification has been derived primarily from the use of cryogenic-actuated valves, the tap-off turbine-drive concept, and catalytic main-chamber ignition for O_2/H_2 systems.

Using cryogenic propellants to actuate valves eliminates the requirement for a gas system for valve actuation; it also promotes physical integration of the components.

The tap-off turbine-drive concept eliminates the conventional gas-generator system. A novel start-system concept for use with a multiple-start storable-propellant system has been evolved. This start-system uses stored tap-off gases to spin the turbine(s) for the next start.

The use of a catalytically-ignited mixture of gaseous oxygen and hydrogen for main-chamber ignition effects a significant simplification for multiple-start O_2/H_2 systems by reducing the ignition system to a single passive component, the catalyst bed.

A number of concepts having a high-chamber-pressure capability, but using fewer pumps than a conventional system (or no pumps) was investigated. The most promising of these was the pulsing engine which is a pressure-fed, high chamber-pressure, low tank-pressure, intermittent-combustion engine. The work on the other concepts of this type is briefly summarized below:

(1) Pressure-Fed/Pump-Fed System - -
In this concept one propellant is pressure-fed while the other is pumped using a conventional turbopump. It was considered for an F_2/H_2 system

wherein the F_2 was pressure-fed. This has a reliability advantage over an all pump-fed system, and a small payload advantage over an all pressure-fed system.

(2) Gas-Drive Jet-Pump - -

This was considered for pumping fluorine. Inefficient separation of the drive gas and pumped propellant resulted in a performance degradation that made the concept undesirable.

(3) Jet-Pump-Fed/Pump-Fed System - -

This concept was considered for an F_2/H_2 system, and used a condensing jet-pump (the drive fluid is a vapor that condenses and becomes a part of the pumped fluid — so no separation is required) for the fluorine. It could be more reliable than an all pump-fed system. A parametric payload comparison with an all pressure-fed system has been made.

(4) Shuttle-Feed Systems - -

Several of these systems were evaluated to determine feasibility and cycle rates. The concept is a high chamber-pressure pressure-fed

O_2/H_2 system wherein propellants are expelled (alternately) from two (for each propellant) small high-pressure tanks by pressurizing with catalytically ignited O_2/H_2. Some of the systems proved to be infeasible. The feasible one required excessively large lines and valves to achieve reasonable cycle rates.

A number of physical integration concepts was evolved. The more notable of these were highly integrated concepts using cryogenic-actuated valves, and the use of a multiple-poppet valve concept to reduce feed-system size for large-thrust engines. Design effort on other concepts is briefly summarized below:

(1) Propellant-Valve/Turbopump Concepts --
Two concepts for integration of main propellant valves and turbopumps were evolved. Both concepts utilize a number of small valves integrated with the pump housing. These concepts can effect size and weight savings for large-thrust engines.

(2) Cartridge Integration Concept --
The cartridge concept essentially consists of designing components so they can be "plugged" into a hole. These holes are parts of a common manifold

and structural member which has built-in passages for the propellant flow. Thus, this concept tends to reduce system size and weight by having components "share" structural elements. Use of this concept is shown for valves and turbopumps.

(3) Start-System Integration -
Two highly-integrated start-system concepts have been evolved for use in a multiple-start NTO/50-50 spacecraft engine.

(4) Three-Leg Gimbal Concept -
The three-leg gimbal concept integrates the gimbal bearing, thrust structure, and propellant inlet ducts. This places the gimbal point closer to the engine center of gravity, thus reducing the required actuator loads.

(5) Tubular Spherical-Combustor -
The tubular spherical-combustor concept is similar to the toroidal-combustor concept (Ref. 12). For low-thrust applications this concept may be easier to cool and fabricate than a comparable toroidal combustor, and may possess a weight advantage.
An investigation has been made of a scheme whereby this concept could be used as a thrust-vector-control device, with a fixed thrust chamber. The combustor would be gimbaled while the thrust chamber remains fixed.

APPROACH

The general method of approach was to first minimize system complexity through functional analysis and integration, and then to investigate methods of physically integrating the required components.

The functional analysis was followed by a rating of the defined "functionally-simple" systems to determine which were "optimum" with respect to operational simplicity, weight, and reliability. These "optima" were then designated as basic systems, and subsequent evaluations of various component and subsystem concepts were based largely on their use in these systems.

Two types of investigations and evaluations of concepts were made, namely, analytical and design.

The analytical investigations were primarily directed at defining and evaluating, (1) novel methods of functionally integrating somewhat conventional components, and (2) novel methods of accomplishing the same function as the basic systems, but with fewer and/or less complex components. This latter effort consisted largely of evaluating methods of obtaining a high-chamber-pressure capability without using turbopumps or increasing propellant-tank pressure.

The design investigations were primarily directed at defining novel methods of physically integrating the components of the basic systems. This generally produced some feedback to the basic systems, and the basic systems were modified as these investigations indicated how further functional simplification could be achieved.

Subsequently preliminary-design layouts were made of the more-promising component and/or system concepts.

SELECTION OF BASIC SYSTEMS

The basic systems are those functional configurations selected as the bases for the various component and sub-system investigations. Many of these systems use turbopumps and, for the most part, somewhat conventional components. Selection is based on ratings for operational simplicity, weight, and reliability.

The selection process consisted of the following:

(1) Compilation of data on existing engine systems,

(2) Definition of reliability guides for reliable configurations,

(3) Functional analysis to determine how component functions could be combined for system simplification,

(4) Concept definition, and

(5) Rating and selection of the basic configurations.

DATA COMPILATION

The format chosen for the compilation and categorization of functional data on somewhat conventional systems was the "morphological" chart, Fig. 1 . Using this chart, one can define virtually all the possible functional configurations that can be formed using essentially conventional components.

To define a concept one starts from the top of the page and follows any one of the numerous possible paths to the bottom of the page, listing the

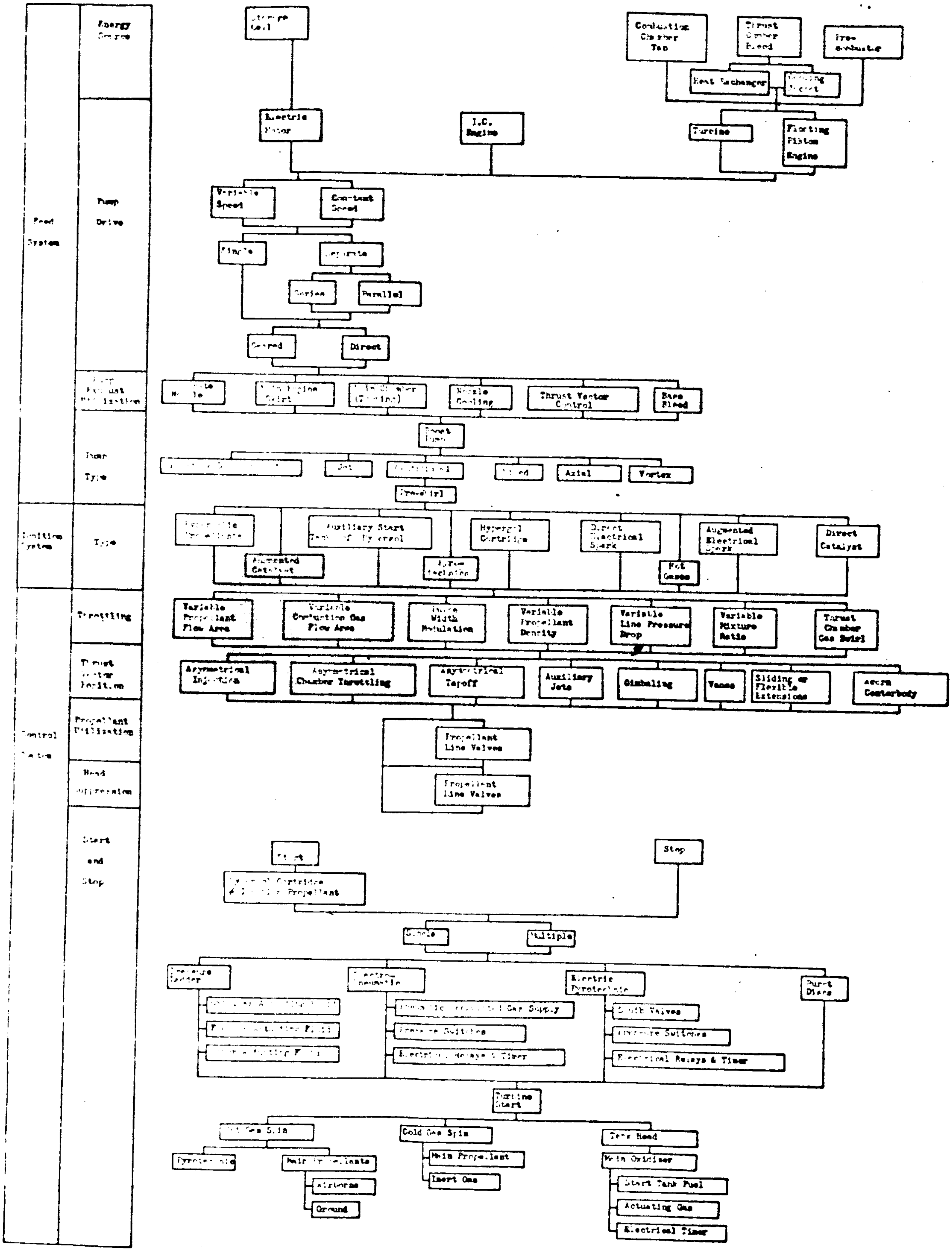

Figure 1. Morphological Chart.

components that are in the blocks which lie on the path followed. For example, the following is a functional description of a system defined using this chart:

(1) Combustion chamber tap-off

(2) Single, constant-speed, direct-drive turbine

(3) Separate nozzle for pump-drive exhaust

(4) Centrifugal pumps

(5) Hypergolic propellants

(6) No throttling

(7) Single start

(8) Pressure-ladder valve sequencing with fuel as actuating fluid

(9) Pyrotechnic hot-gas-spin turbine-start (solid spinner)

In this manner different configurations can be formed and evaluated; the use of this chart for conventional systems is discussed in more detail below in the discussion of functional analysis.

RELIABILITY GUIDES

A compilation of historical reliability data was made to serve as a general reliability guide during the preliminary stages of concept definition. Guides were established for four categories, namely

(1) Turbopumps

(2) Turbine-Drive and Start Systems

(3) Throttling Methods

(4) Ignition Systems

Turbopumps

A brief description of common turbopump failure-modes is given, and possible remedies for each mode are suggested.

Leakage past the primary shaft-seals is a frequent occurrence. This problem is more prevalent in cryogenic systems. The greatest reliability gain in this area can be achieved by employing a double-seal design with adequate provisions for drainage between the seals. Reduction of fluid pressure at the seal face, by suitable choice of various clearances, can also improve reliability in this area.

Bearing-lubrication failure results from an inadequate flow of lubricant. Furthermore, inadequate control of lubricant flow can, in the case of excessive flow, lead to a failure-producing pressure-buildup in the pump casing. Therefore, special care must be taken to assure lubricant flow is within acceptable limits.

Pump overspeed, resulting from cavitation, can result in catastrophic turbopump failure. Reliability in this area can be improved by adequate control of pump NPSH or by use of an overspeed detection device.

System contamination is a cause of numerous failures. The frequency of this mode of failure can be reduced by the proper use of filters. Care in assembly, and rigid inspection are also helpful in this regard.

Human error contributes to many malfunctions. This mode can be reduced by employing designs that minimize the human role in assembly and utilization of the turbopump, and by careful inspection.

Turbine-Drive and Start Systems

A list of some possible combinations of turbine-drive and start systems is presented; these systems are ranked with regard to reliability. The original data were generated for a thrust level of 50K using LO_2/LH_2 as propellants, and were based on 6 engine starts. This is followed by a brief general discussion of start-system reliability.

The ranking presented below should be a good guide for LF_2/LH_2 systems, and a fair guide for N_2O_4/N_2H_4-UDMH(50-50) systems. The system combinations are listed below in order of decreasing reliability.

Turbine Drive	Start
(1) Thrust-chamber cooling-jacket tap-off	Gas spin
(2) Thrust-chamber tap-off	Gas spin
(3) Thrust-chamber tap-off	Start tank

(4)	Thrust chamber cooling-jacket tap-off	Monopropellant gas-generator
(5)	Bipropellant gas-generator	Tank head
(6)	Bipropellant gas-generator	Start tank
(7)	Bipropellant gas-generator	Gas spin
(8)	Bipropellant gas-generator	Monopropellant gas-generator
(9)	Thrust chamber tap-off	Solid-propellant spinner
(10)	Dual gas-generators	Start tank
(11)	Thrust chamber cooling-jacket tap-off	Solid-propellant spinner
(12)	Bipropellant gas-generator	Solid-propellant spinner

Selection of a start system for any particular engine, especially where restart capability is involved, may (of necessity) be influenced to a great extent by other system features. From a reliability standpoint, however, the optimum system, when feasible, is tank-head start. Normally, it requires only components that are already in the engine and/or vehicle system. Pressure-fed systems are naturally of this type. However, this method cannot always be used for pump-fed systems, so one of the following systems is frequently used.

The solid-propellant gas-generator (SPGG), also designated turbine spinner, represents the most reliable of these start systems for single-start applications. It possesses maximum simplicity and has proven to be an extremely reliable system in Rocketdyne engines. It is not particularly adaptable to multiple-start systems however, without a significant increase in system complexity and consequent degradation of the inherent high reliability.

In engines where LH_2 is used as the fuel, spin-start of the turbine using expanding gaseous-hydrogen is highly reliable, particularly where multiple-starts are required. Its performance in the Rocketdyne J-2 engine has been very satisfactory.

Other start systems, utilizing hot-gas from a gas-generator for initial spin of the turbine, are quite reliable and are within the current state-of-the-art. Many variations can be devised using monopropellant or bipropellant liquids, or hybrid systems for gas-generation. Reliability of these systems can vary substantially depending on system complexity and propellants. The start-systems of systems 301, 302, 303, and 304 of this report illustrate this type of application. These systems, like the gaseous-hydrogen spin-start, are adaptable to multiple-start systems.

Throttling Methods

In considering throttling of a liquid-propellant engine, the more feasible methods fall into 3 general categories:

(1) Flow control upstream of injector

(2) Injection area control

(3) Propellant density control

Each of these throttling systems must be provided with the proper signal or signals to operate through a servo-system or other means dependent on the degree of throttling required. For the purposes of this discussion, we can consider this function to be external to the control system under study. For a particular throttling range or control requirement, the reliability of the system required to provide these signals would be essentially the same for all the systems.

In addition to this, a system of valves, actuators, gas-storage bottles, etc. is required, dependent on the system under consideration; these are the components that will be considered in this discussion of throttling-system reliability.

Six basic systems will be considered; these are:

(1) Momentum exchange

(2) Inert gas injection (in main lines)

(3) Main-line valve with inert-gas injection

(4) Main-line throttling-valve

(5) Movable-pintle injector

(6) Hot-gas injection (in main lines)

These systems are compatible with either a pressure-fed or a pump-fed system. In addition, the pump-fed system presents another feasible method for flow control upstream of the injector by control of pump speed as indicated in proposed systems 102 and 202.

For optimum throttling-system reliability, the control system offering the smallest increase in complexity and/or number of significantly unreliable components is, generally speaking, the one to be selected. For the above systems, this is represented by:

(a) Control of pump speed (applicable to pump-fed only)

(b) Simple momentum exchange (see below)

(c) Main-line throttling-valve

(d) Inert-gas injection in main lines

(e) Hot-gas injection in main lines

Where these systems are adequate for the required degree of thrust-level control, they provide the minimum decrease in overall system reliability. Where increased throttling range or more rigid throttling control is required to perform the designated mission, the complexity of the throttling system necessarily increases---with a resultant decrease in system reliability.

The momentum exchange system ((b) above) serves as a good example of this factor. The simple momentum-exchange system, consisting of 2 on/off valves and 2 throttle valves in conjunction with a momentum-exchange injector (a relatively simple injector) presents an extremely reliable system within certain limits of thrust-level control. Increased throttling requires the addition of 2 more throttle valves and 2 gas-injection valves to the system, consequently decreasing reliability. If deeper throttling or more rigid control is required, an optimum momentum-exchange system utilizing control of mixture-ratio in addition to the above-described system is necessary. The increased complexity of this system further degrades reliability, but for optimum control of thrust level, the trade-off must be made between reliability and performance and, if applicable, other parameters such as weight, cost, etc.

Main-line valve with inert-gas injection is another example of increased complexity (being a combination of (c) and (d) above), resulting in reliability degradation, but providing deeper throttling and closer control of thrust-level than either (c) or (d).

The movable-pintle injector presents the least reliable of the aforementioned systems. This is due to the increased complexity of the control system required for its operation.

Ignition Systems

Discussions are presented for 5 classes of ignition systems. These discussions are based on current, feasible state-of-the-art.

From the standpoint of reliability, the optimum ignition system is one in which the system propellants are hypergolic, or self-igniting on contact. However, system performance or other considerations frequently rule out this possibility, and an ignition device must be provided.

The most reliable system of this type is a hypergolic slug, sequenced to precede one of the system propellants into the combustion chamber, thus assuring ignition upon contact with the other propellant. This method is widely used on current Rocketdyne engines and has proven extremely reliable. It is, however, primarily adaptable to single-start systems. Its use in multiple-start applications results in increased complexity and a consequent degradation of the high inherent reliability.

A somewhat similar ignition method which is more readily adaptable to multiple-start systems is catalytic ignition. This provides a greater overall system reliability than the more complex hypergol system. In this analysis, systems 101 and 102 are representative of this choice, and as such possess optimum ignition-system reliability potential.

Other ignition methods are the use of an Augmented Spark Igniter (ASI) system (consisting of spark exciter, monitor, spark plugs, and associated wiring, etc.), or the use of spark plugs only to provide propellant ignition. The A.S.I. system can be developed to a very high reliability level by the use of redundancy, consisting of dual exciters, dual monitors, and dual plugs. This redundant A.S.I. system is particularly adaptable to a multiple-start system.

Another ignition method considered is the use of pyrotechnic igniters; these have been used extensively in various systems, but have proven somewhat less reliable than the other systems discussed. Here again, these igniters are not readily adaptable to multiple-start systems.

FUNCTIONAL ANALYSIS

Functional analysis for the basic systems was initiated by defining the general system requirements. For spacecraft this meant systems that were throttleable and restartable (Ref. 1), for boosters non-throttleable, single-start systems were to be considered.

Starting with the above specifications, effort was directed at defining the nominal feed-system configurations, that is, systems that were known to be feasible for each propellant combination, and which were reasonably representative of optimum conventional systems. Based on this evaluation, the feed system characteristics listed in Table 1 were selected.

TABLE 1

SPACECRAFT (THRUST = 40K)

	Propellants	*Fuel Pump	*Oxid. Pump	Comments
(1)	LO_2/LH_2	Centrifugal	Centrifugal	Direct-drive turbines with pumps on separate shafts
(2)	LF_2/LH_2	Centrifugal	Centrifugal	Direct-drive turbines with pumps on separate shafts
(3)	N_2O_4/N_2H_4-UDMH(50-50)	Centrifugal	Centrifugal	Direct-drive single turbine with pumps on a common shaft

BOOSTER (THRUST = 6M)

	Propellants	*Fuel Pump	*Oxid. Pump	Comments
(4)	LO_2/LH_2	Axial	Centrifugal	Direct-drive turbines with pumps on separate shafts
(5)	LO_2/RP-1	Centrifugal	Centrifugal	Direct-drive single turbine with pumps on a common shaft

*Pumps used in throttleable systems will utilize pre-whirl to minimize NPSH requirements during throttling (Ref. 2).

For purposes of initial concept comparison, it is assumed that multiple fuel or oxidizer pumps are not used. It should be noted that the optimum number of pumps for a 6M pound thrust engine has not been established; however, a study to define this number is currently being conducted at Rocketdyne as part of another program.

Review of the possible systems which could be defined using Fig. 1 and the data presented in the Reliability Section, indicated that use of the following functional concepts would tend to reduce system complexity, and improve reliability:

(1) Combustion-chamber tap-off

(2) Hypergolic main propellants

(3) Pressure-ladder for valve sequencing--using fuel and/or oxidizer for valve actuation

(4) Gas-spin start (for restartable systems)

(5) Hot-gas spin, solid spinner (for single start systems)

(6) Variable line-pressure-drop for throttlable systems (throttle-valve in the tap-off line)

The applicability of all of these concepts to the propellants being considered in this program is not obvious. In particular, two questions have been raised during the course of this investigation:

(1) Can cryogenic fluids effect acceptable valve actuation? (For use in pressure-ladder sequencing)

(2) Can the LO_2/RP-1 and LO_2/LH_2 propellant combinations be made hypergolic without gross modifications?

An investigation of the possibility of actuating valves with cryogenic fluids has indicated that the primary potential problem area is associated with phase changes of the actuating fluid during valve actuation. This could cause irregularities in valve-actuation times. Experimental work (done at Rocketdyne early in the F-1 program) indicated acceptable valve-actuation could be achieved using a cryogenic fluid. In addition, Rocketdyne's J-2 engine currently uses liquid oxygen to actuate the control valve for the main oxidizer valve; operation of this valve has been satisfactory. Based on these data, it has been assumed that cryogenic fluids can be successfully used for valve actuation.

Two methods of making the LO_2/LH_2 propellant combination hypergolic are currently being considered in the industry: (1) Addition of fluorine to the oxygen, and (2) Addition of ozone fluoride (O_3F_2) to the oxygen. Recent experiments (Ref. 3) indicate that approximately 35% LF_2 added to LO_2 is required to make this combination hypergolic. Data on O_3F_2 indicate 0.05% is sufficient for hypergolicity. No further consideration is given to these additives. It is felt that to consider the addition of 35% LF_2 to the LO_2 is outside the scope of this program, since it would constitute a major change in propellants. The use of O_2F_2 is not considered to be compatible with the program objective of operational simplicity because of the logistics problems that could arise from its use. These difficulties stem from two properties of O_3F_2, namely, its approximately 18 day half-life and its tendency to decompose at temperatures slightly higher than LO_2 temperatures.

Furthermore, insofar as system concepts are concerned, should LO_2/LH_2 be made hypergolic by use of some additive, it could then use concepts evolved for the naturally hypergolic combination, LF_2/LH_2. Thus no promising concept is neglected simply because it is peculiar to hypergolic systems.

Work at Rocketdyne indicates that O_2/RP-1 can be made hypergolic by the addition of 5% to 6% of fluorine to the oxygen. The gain in simplicity for most O_2/RP-1 systems (single-start booster engines) is probably not justified, since an ignition failure for such a booster would not be catastrophic.

As a part of this investigation of ignition-system simplification (with emphasis on hypergolicity) the use of an ignition device utilizing the principle of catalytic ignition of oxygen and hydrogen was considered. This device could consist of an ignition chamber and a catalyst bed. Oxygen and hydrogen burning in the small ignition chamber would be used to ignite the main propellants. Considerable work has been done at Rocketdyne with such devices using research-type hardware. More recent experience includes design and test of such a device for the J-2 engine. Indications are that this type of ignition system can be designed to function with high reliability. For this reason, its feasibility is presumed for the purposes of this study.

CONCEPT DEFINITION

In the paragraphs that follow, brief descriptions and explanatory notes will be presented for a number of system concepts that have been defined and rated. The operational simplicity, reliability, and weight ratings of these basic systems are subsequently presented and discussed.

Because the operation of many of the systems presented in Figs. 3 through 20 is similar, complete descriptions will not be given for all the systems.

The primary events in the start sequence for system 101 are as follows:

(1) Electrical start signal opens the normally closed start-tank valve (STV), and closes the normally open oxidizer cut-off valve (OCV)

(2) Gas from the start tank accelerates the turbopumps and pressure starts to buildup in the main lines.

(3) Main fuel valve (MFV) is opened by fuel pressure in the main line; the igniter oxidizer valve (mechanically linked to the main fuel valve) is opened; lines start to prime, and propellants start to burn in the catalytic ignition-chamber.

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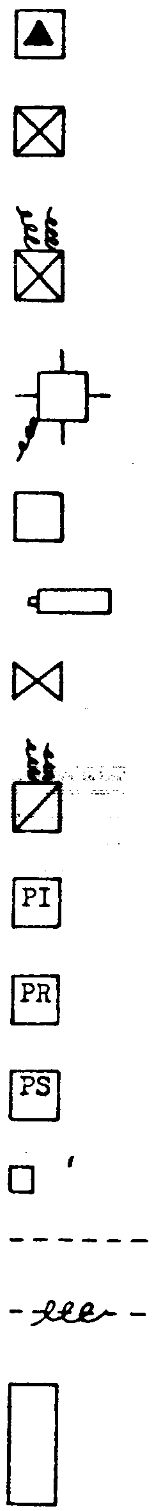
CHECKED BY:	A DIVISION OF NORTH AMERICAN AVIATION INC	REPORT NO.
DATE:	Fig. 2	MODEL NO.
	Schematic Symbols	



Check Valve

Propellant Valve

Solenoid Valve

4-way Solenoid Valve

Hot-gas Valve

Squib Valve

Fill and Drain (or Vent) Valve

Servo-motor

Pressure Intensifier

Pressure Regulator

Pressure Switch

Temperature Detector

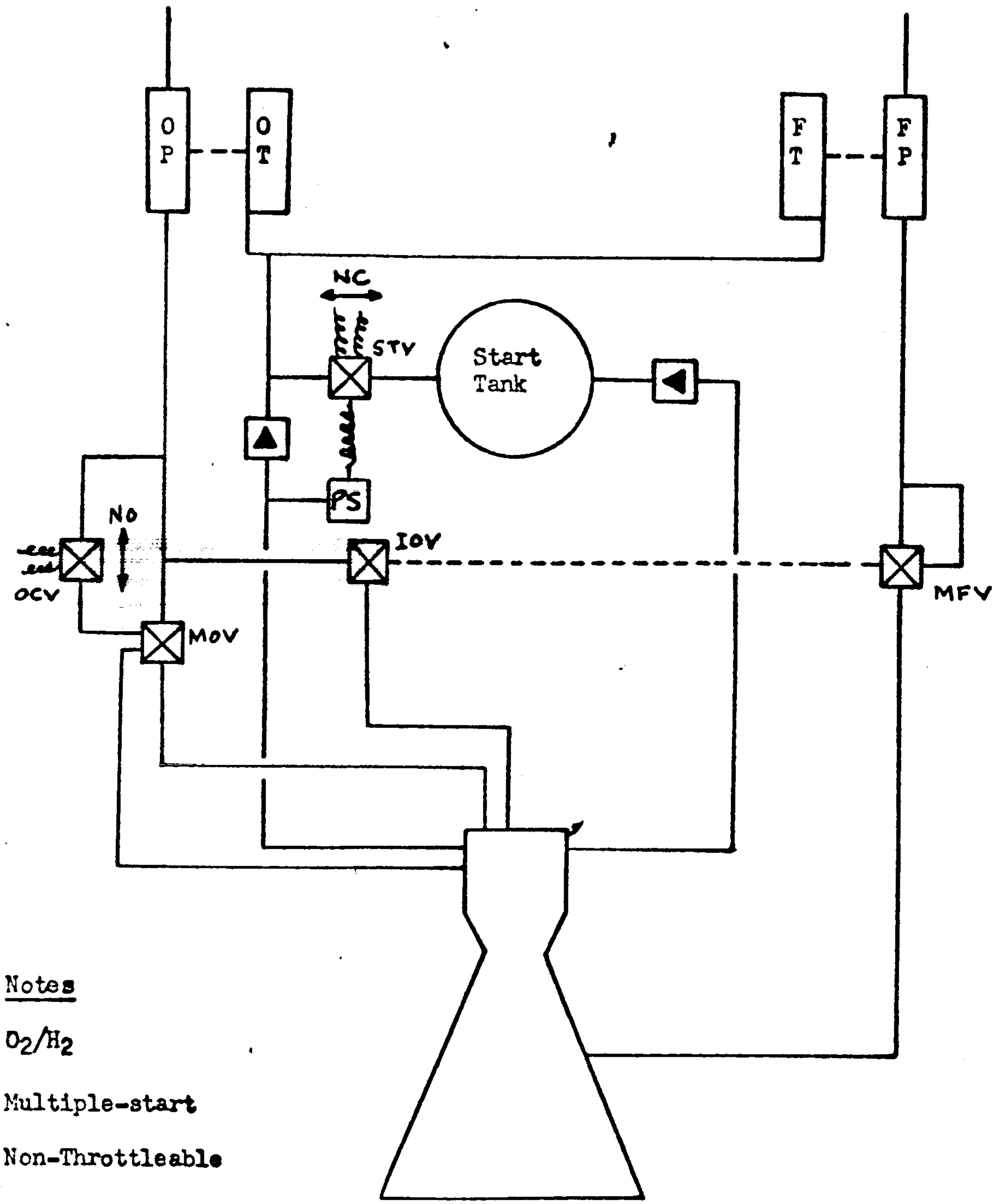
Mechanically Linked

Electrically Linked

OP - Oxidizer Pump
OT - Oxidizer Turbine
FP - Fuel Pump
FT - Fuel Turbine
T - Turbine

Fig. 3

System 101



Notes

O_2/H_2

Multiple-start

Non-Throttleable

(4) Ignition stage is achieved.

(5) Ignition-stage chamber-pressure opens the main oxidizer valve, and chamber pressure starts to rise.

(6) Pressure switch (PS) in the tap-off line is actuated by the pressure from the chamber indicating the system is ready to bootstrap; actuation of this switch (which is electrically linked to the start-tank valve) de-energizes (closes) the start-tank valve (STV).

(7) The system bootstraps, and mainstage ensues.

(8) The start-tank is refilled from the thrust-chamber cooling jacket.

The cut-off sequence is:

(1) Cut-off signal de-energizes two solenoid valves: the oxidizer cut-off valve (OCV), and the start-tank valve; the latter was already de-energized by the pressure switch in the tap-off line, but this additional open switch in the circuit is required to prevent the valve from opening when the pressure switch is deactuated by decaying tap-off pressure.

(2) De-energizing the OCV sends oxidizer to the closing ports of the MOV, thus closing it.

(3) Chamber pressure decays, and the main fuel valve closes; cut-off is effected.

System 102 is identical to system 101 except for the addition of a hot-gas throttling valve in the tap-off line, and a servo motor for actuating the valve.

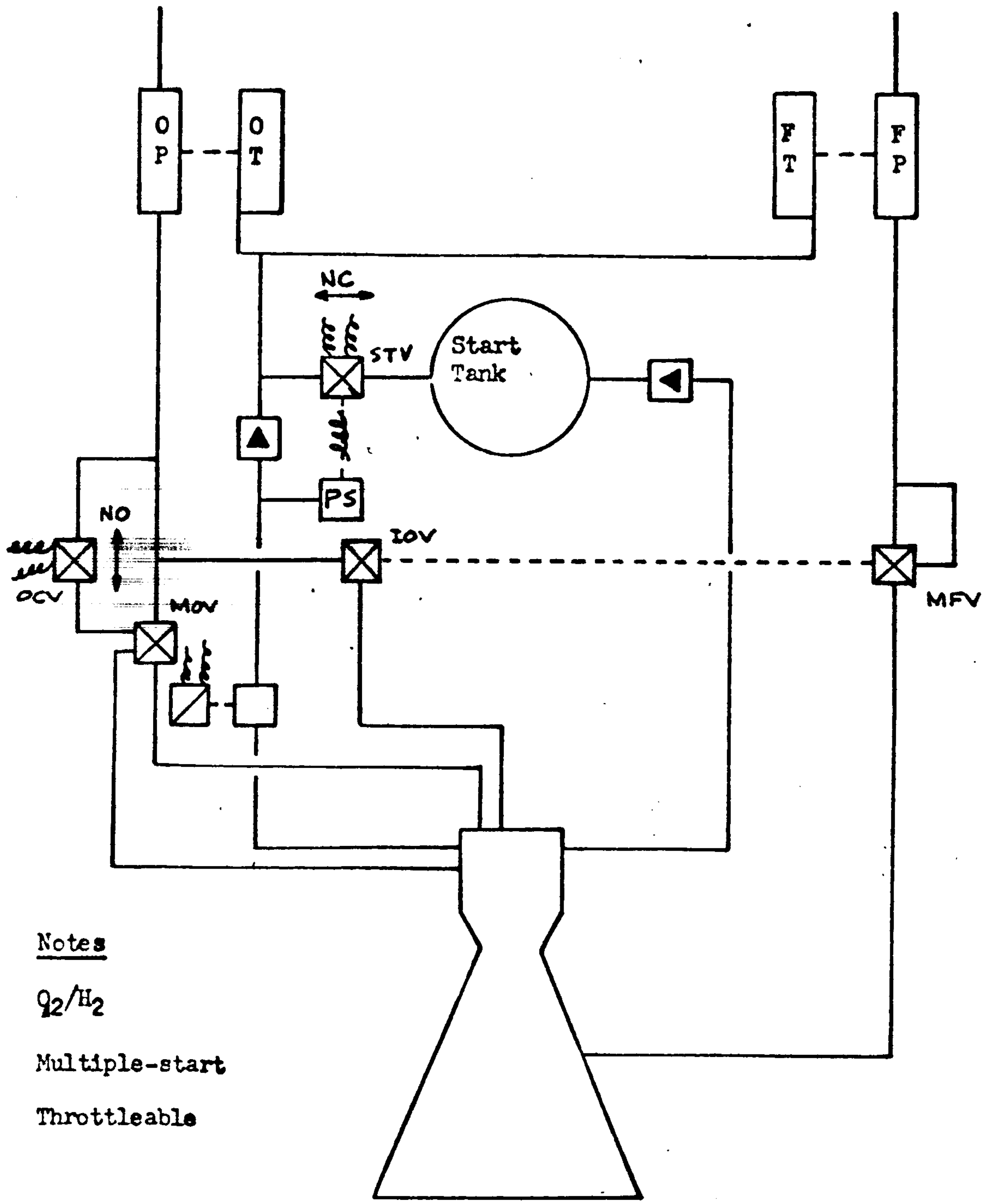
System 103 differs from 101 in that it is a single-start system which uses a solid-propellant spinner (SPGG) for starting, a hypergolic fluid for ignition, and a squib-actuated valve for cutoff. The start signal ignites the solid-spinner, and after the igniter oxidizer-valve (IOV) is opened, oxidizer pressure builds up in the hypergol cartridge until two burst diaphragms are ruptured; hypergol fluid then enters the chamber, and is followed by igniter oxidizer, and ignition stage ensues. The cutoff signal actuates a squib valve (OCV) which allows oxidizer from the main line to go to the closing port of the main oxidizer valve. Decaying fuel-pump discharge pressure closes the main fuel valve, thus effecting cutoff.

System 104 is identical to 101 except for the addition of a reciprocating piston pressure-intensifier (PI) on the start-tank recharging line. This would allow the start tank to be charged to a higher pressure than is available directly from the thrust-chamber cooling-jacket. This could effect reductions in start time, and system size and weight.

System 105 is identical to 104 except that a pressure regulator (PR) is used to regulate the start-tank discharge-pressure. This could improve starting characteristics, and eliminate the need for high-pressure turbine-

Fig. 4

System 102



Notes

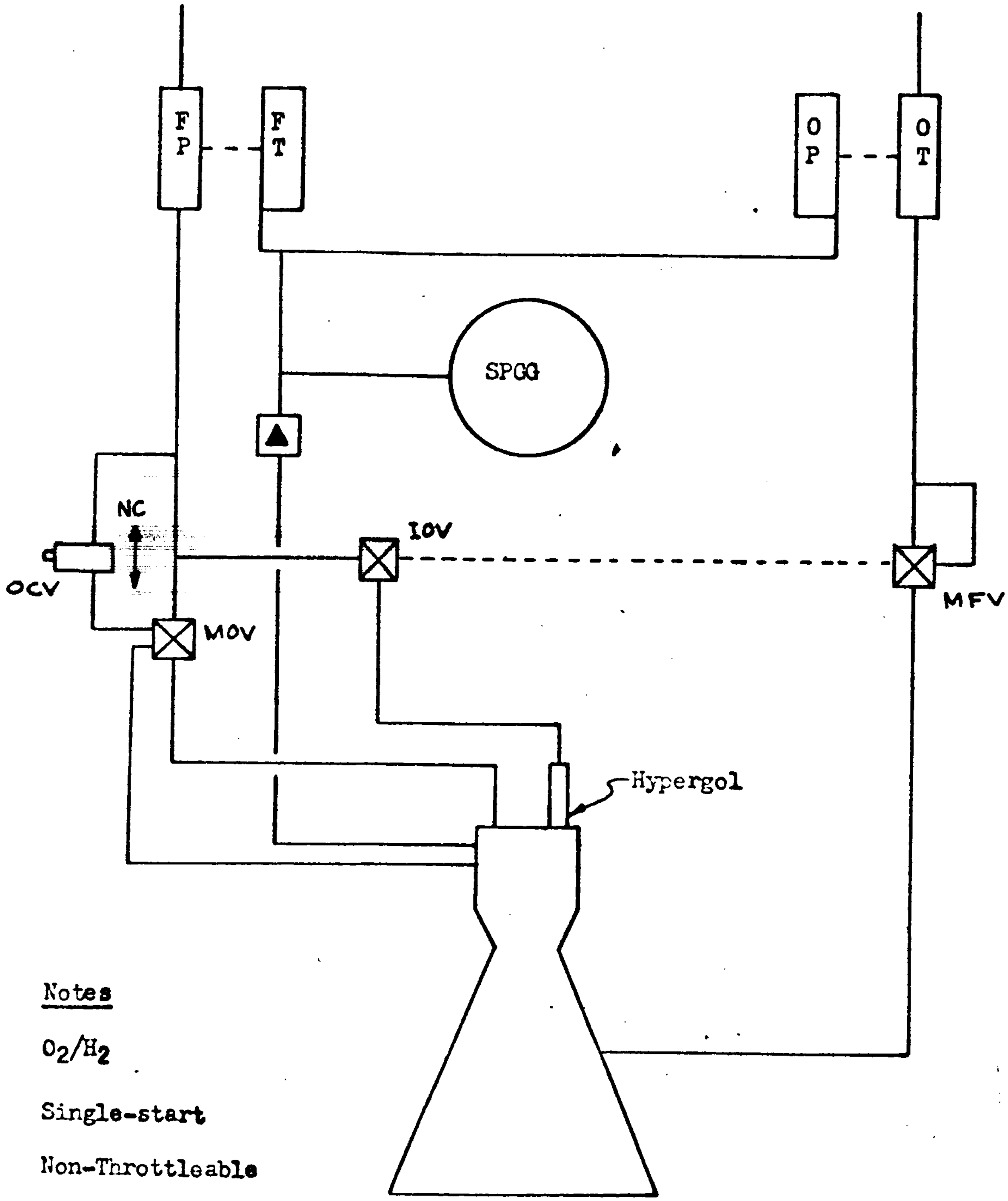
O_2/H_2

Multiple-start

Throttleable

Fig. 5

System 103



Notes

O_2/H_2

Single-start

Non-Throttleable

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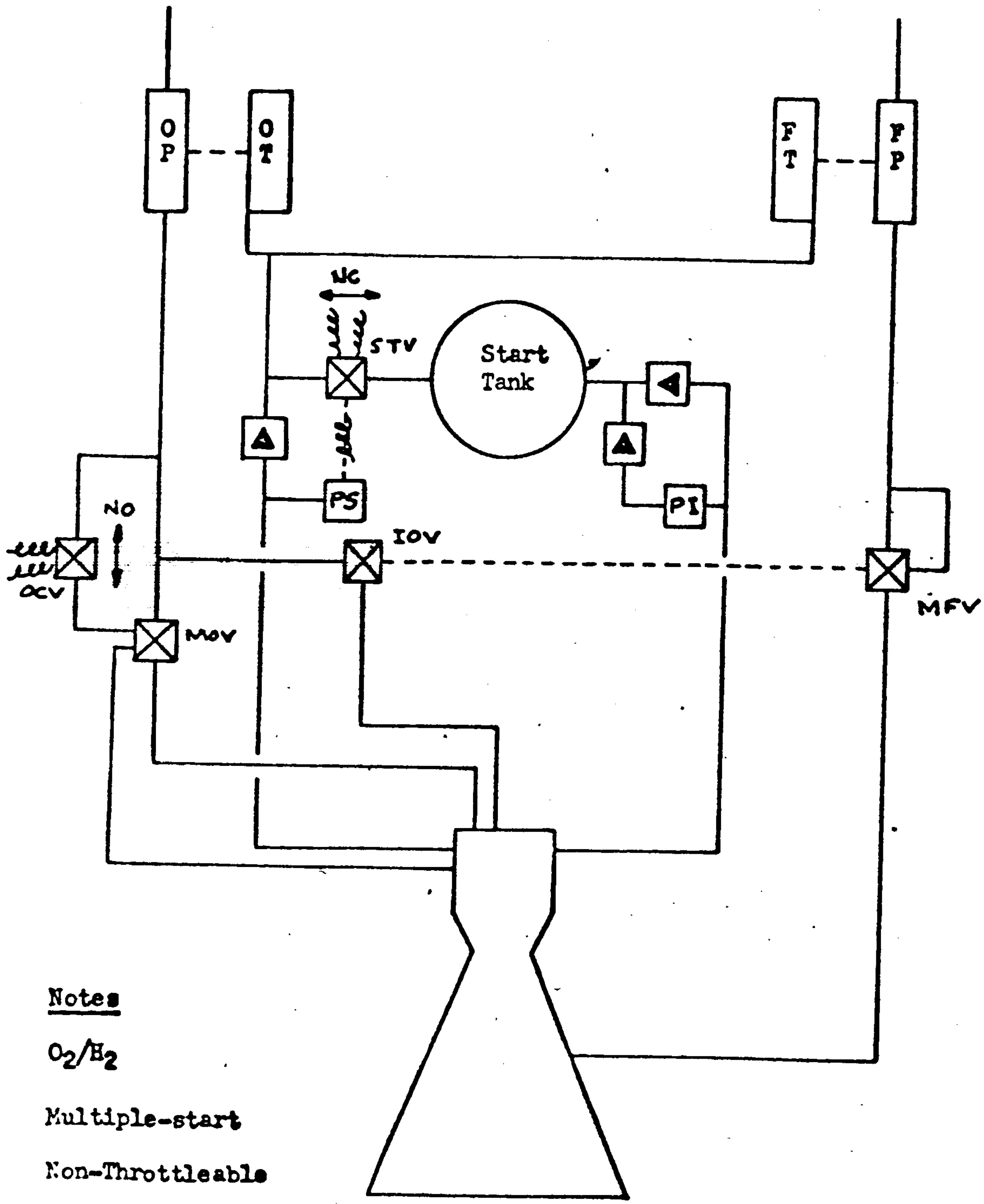
DATE:	System 104	MODEL NO.



Notes

O_2/H_2

Multiple-start

Non-Throttleable

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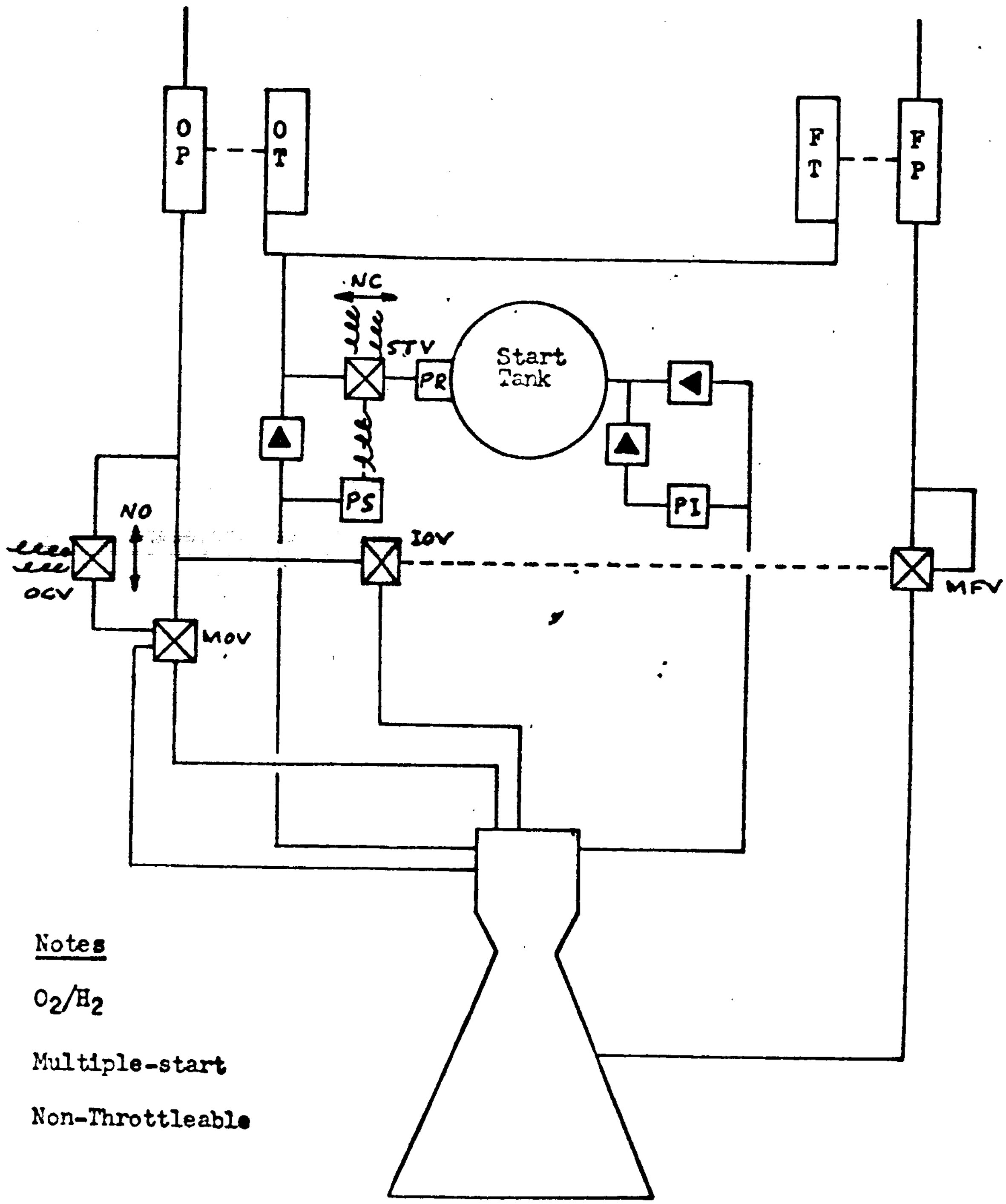

Notes

O_2/H_2

Multiple-start

Non-Throttleable

inlet manifolds, which might otherwise be necessary for systems using a pressure intensifier.

System 201 differs from 101 in that the main oxidizer-valve and main fuel-valve are mechanically linked, and no igniter valve is used. No igniter valve is required because the propellants are hypergolic, and both are cryogenic.

System 202 is identical to 201 except for the addition of a hot-gas throttling valve in the tap-off line.

System 203 differs from 201 in that a pressure intensifier is used in the start-tank recharging line, and a two-stage fuel valve is used. In this concept the fuel valve opens in two steps; the second step of opening is linked to the main-oxidizer valve. The second step cannot occur until an appropriate temperature is sensed at the fuel injector, thus indicating that the fuel system has been primed, and if required, adequately chilled down. This provides a flexibility that could be required for suitable fuel lead at ignition, and pre-ignition chill down.

System 204 is identical to 203 except that a pressure regulator (PR) is used to regulate the start-tank discharge-pressure.

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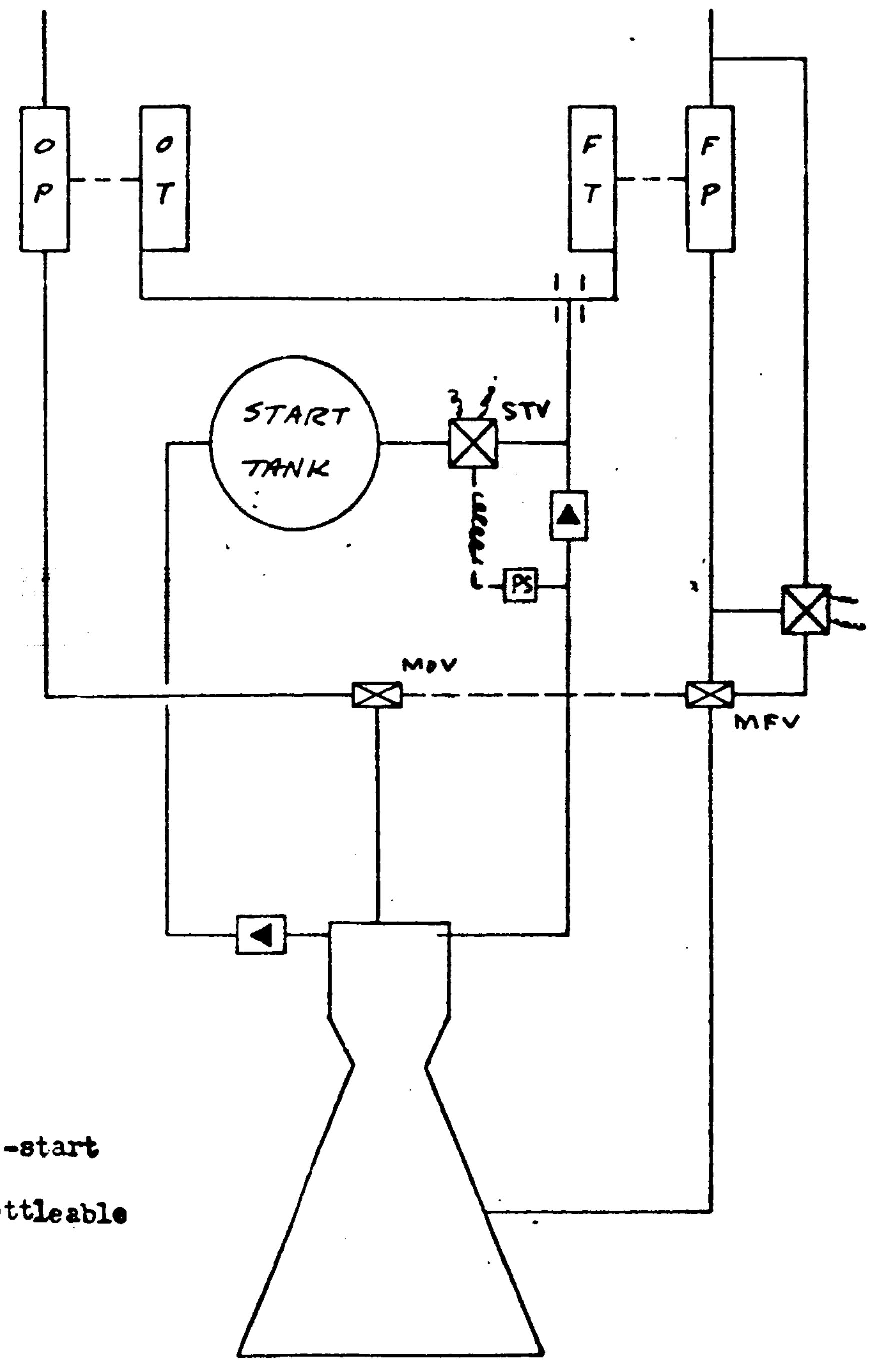


Notes

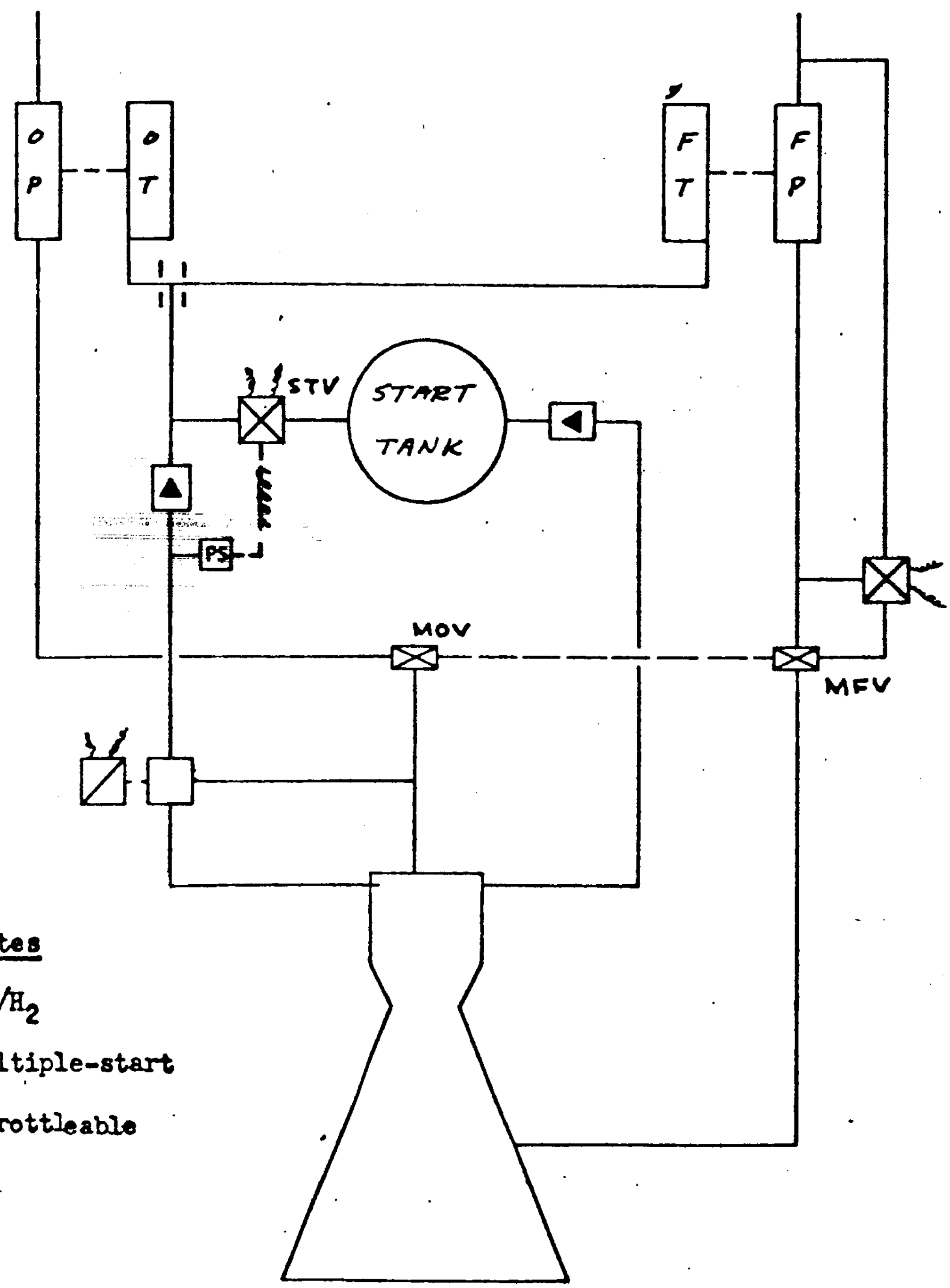
F_2/H_2

Multiple-start

Non-throttleable

Fig. 9

System 202



Notes

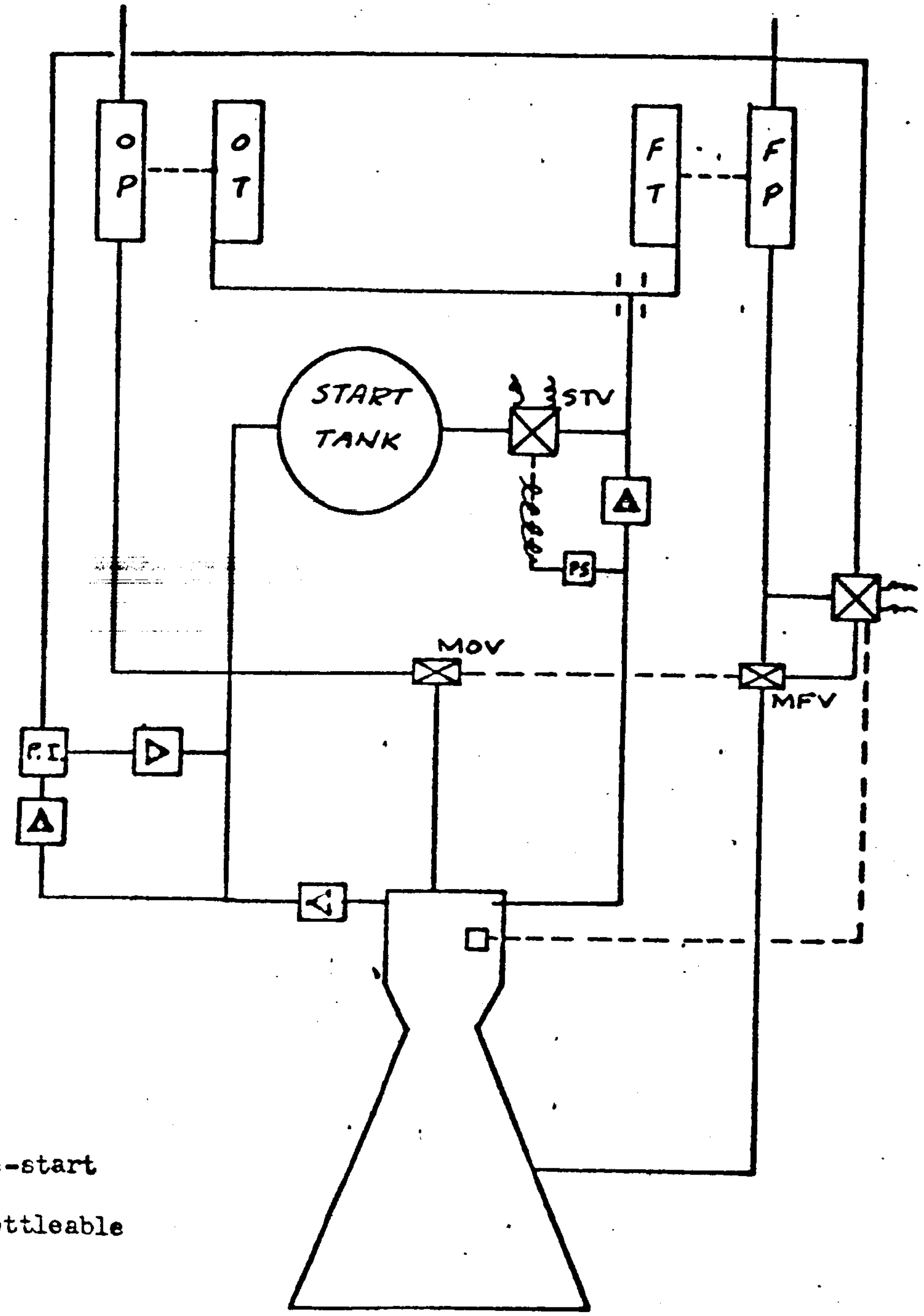
F_2/H_2

Multiple-start

Throttleable

Figure 10

System 203



Notes

F_2/H_2

Multiple-start

Non-Throttleable

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Fig. 11

System 204

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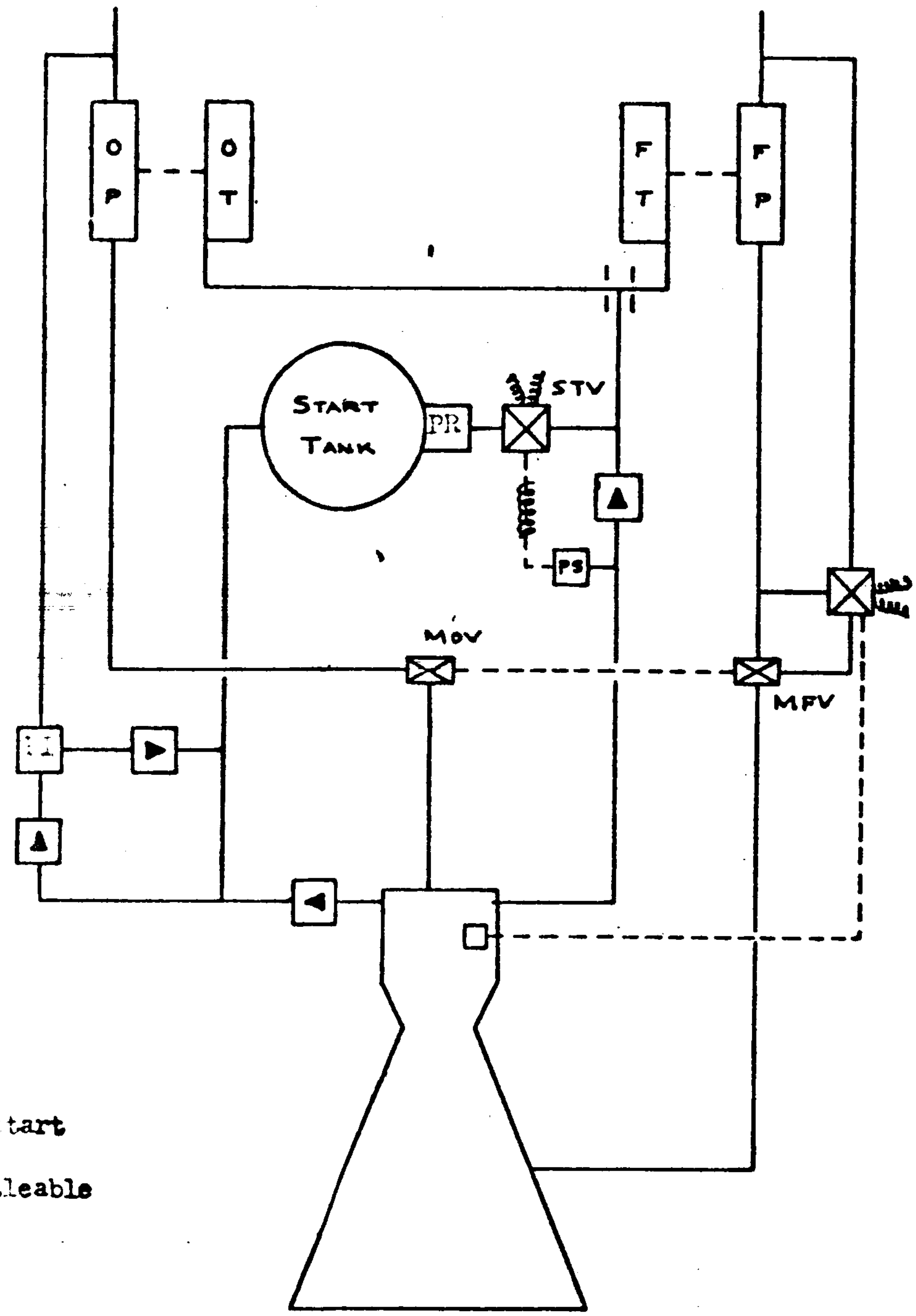
MODEL NO



Notes

F_2/H_2

Multiple-start

Non-Throttleable

System 205 differs from 201 in that it has no oxidizer turbopump. It utilizes a condensing jet-pump (see section on concept evaluation) to pump the oxidizer. A gas generator for producing the jet-pump drive-fluid is not shown.

The primary events in the start sequence for system 301 are:

(1) Electrical signal opens the normally closed start-tank valve (STV) and the normally closed (NC) flow paths of the main control-valve (MCV),

(2) Start propellant enters the gas generator and start to decompose,

(3) Gas-generator gas accelerates the turbopump and pressure starts to build up in the main lines,

(4) The main fuel valve (MFV) is opened by fuel-pump discharge pressure, and oxidizer igniter flow passes through the MCV on its way to the thrust chamber,

(5) Fuel flow and igniter-oxidizer flow enter the thrust chamber, and ignition is established,

(6) Ignition-stage chamber-pressure (sensed through the fuel-inlet manifold) opens the main oxidizer valve, and chamber pressure starts to rise,

(7) Pressure switch (PS) in the tap-off line is actuated by the pressure from the chamber indicating the system is ready to

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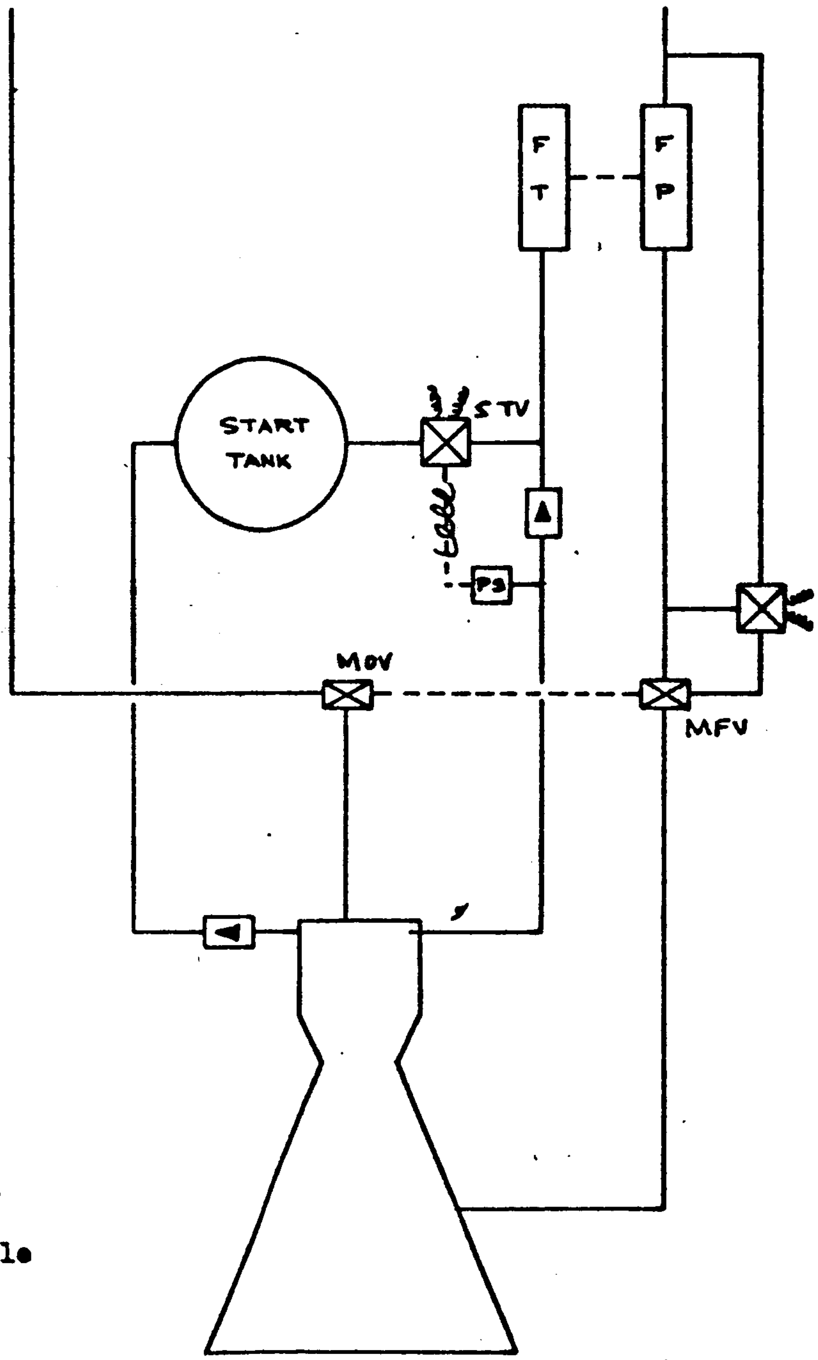
DATE	System 205	MODEL NO



Notes

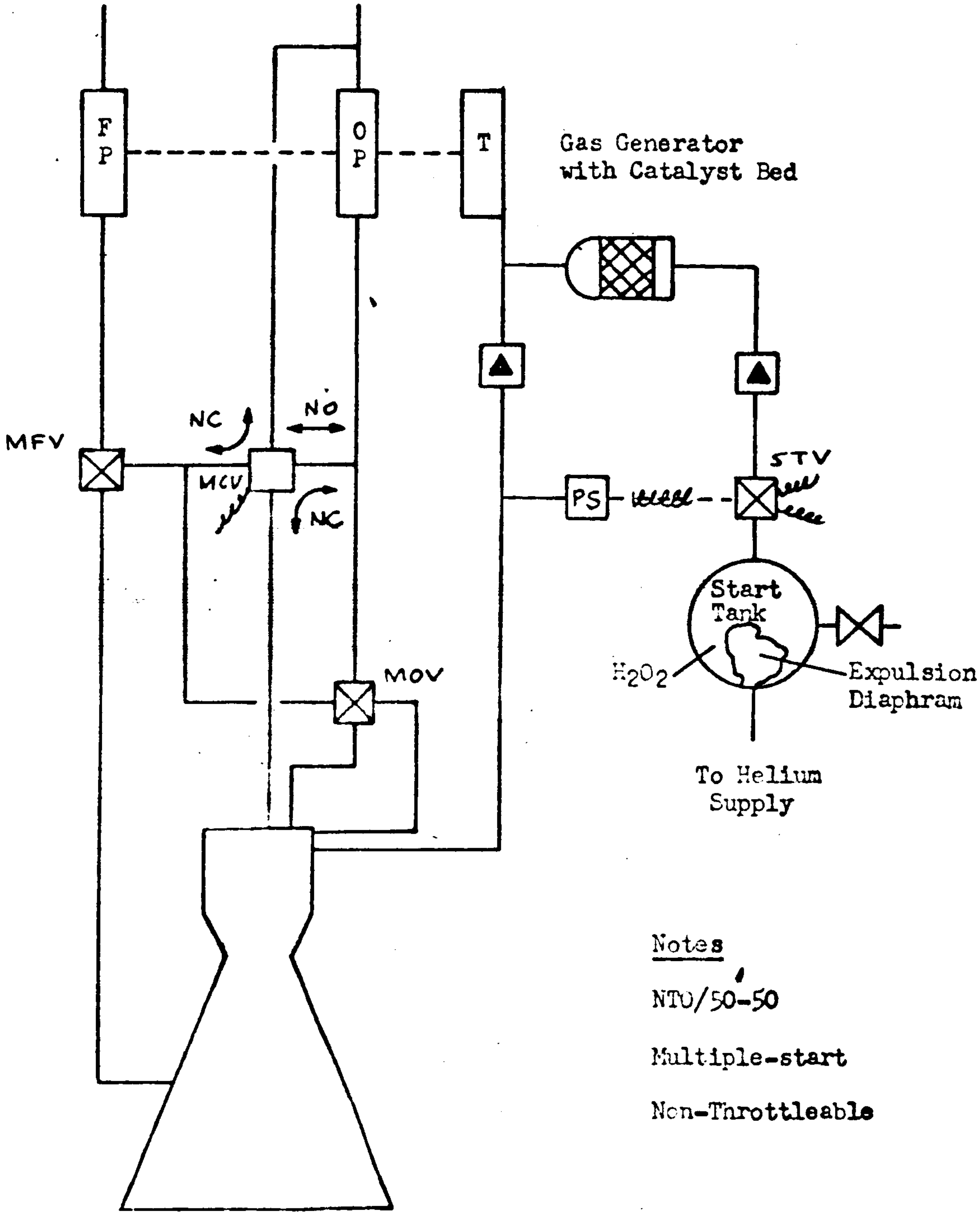
F_2/H_2

Multiple-start

Non-Throttleable

Fig. 13

System 301



Notes

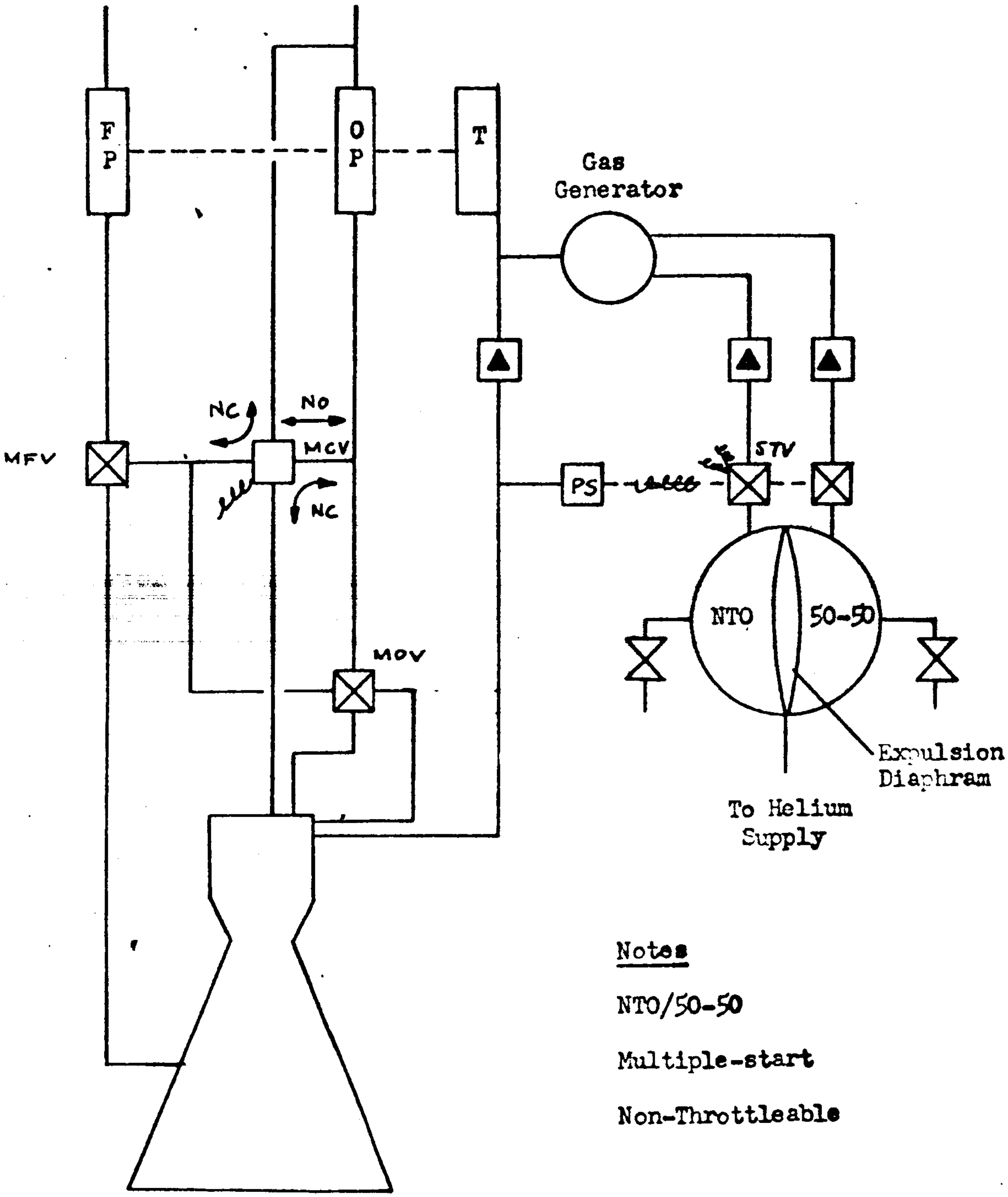
NTO/50-50

Multiple-start

Non-Throttleable

Fig. 14

System 302



Notes

NTO/50-50

Multiple-start

Non-Throttleable

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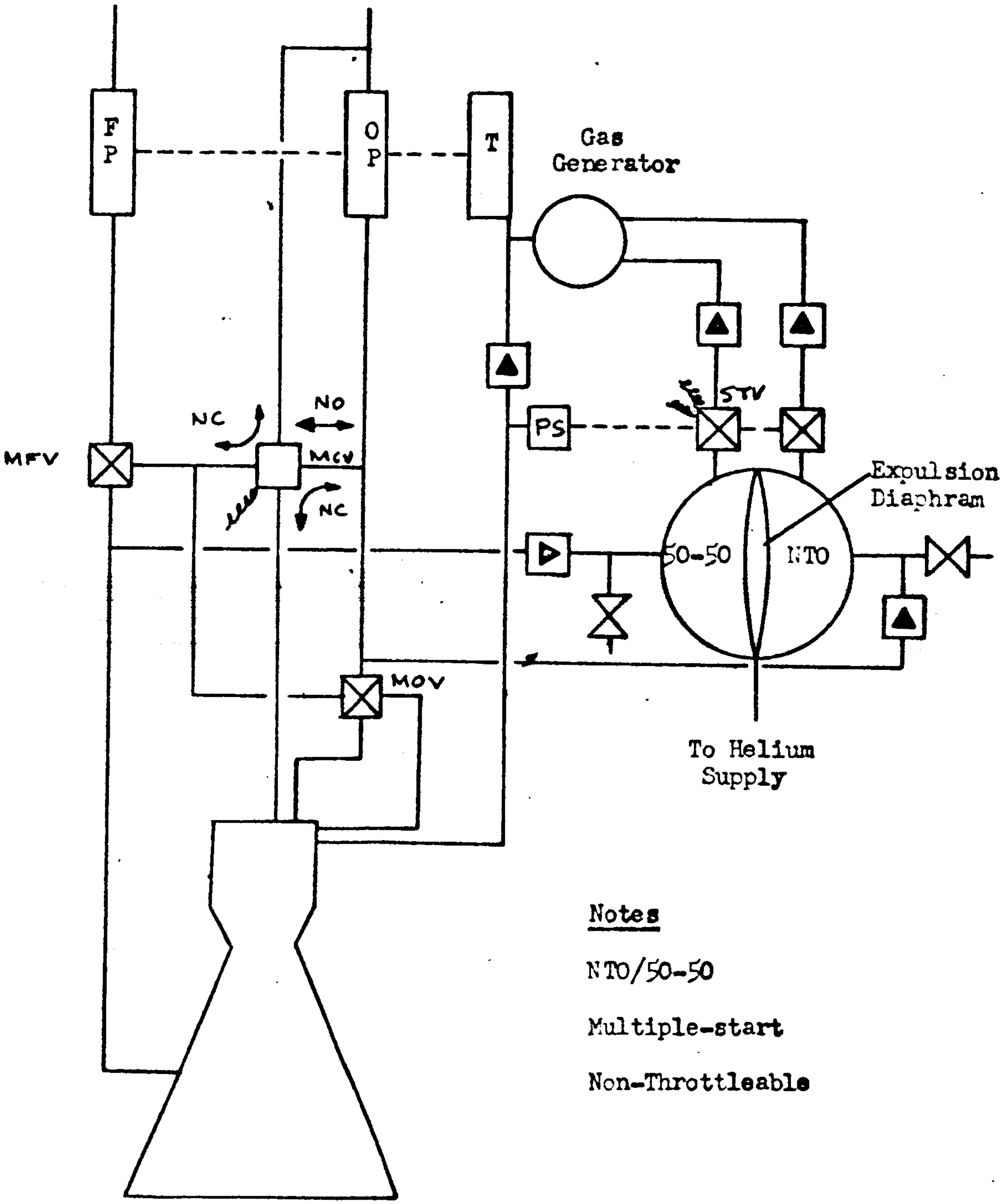

Notes

NTO/50-50

Multiple-start

Non-Throttleable

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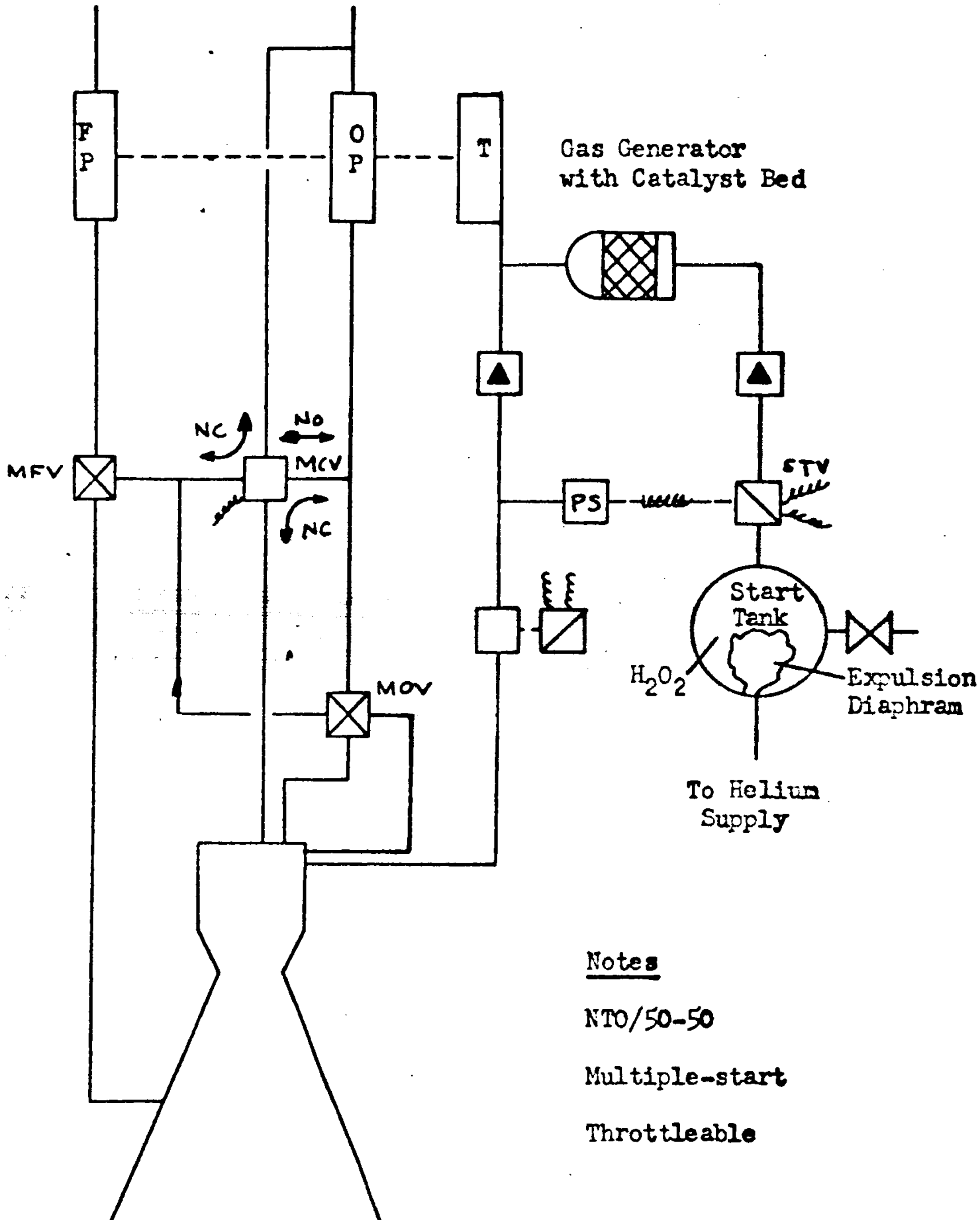


Notes

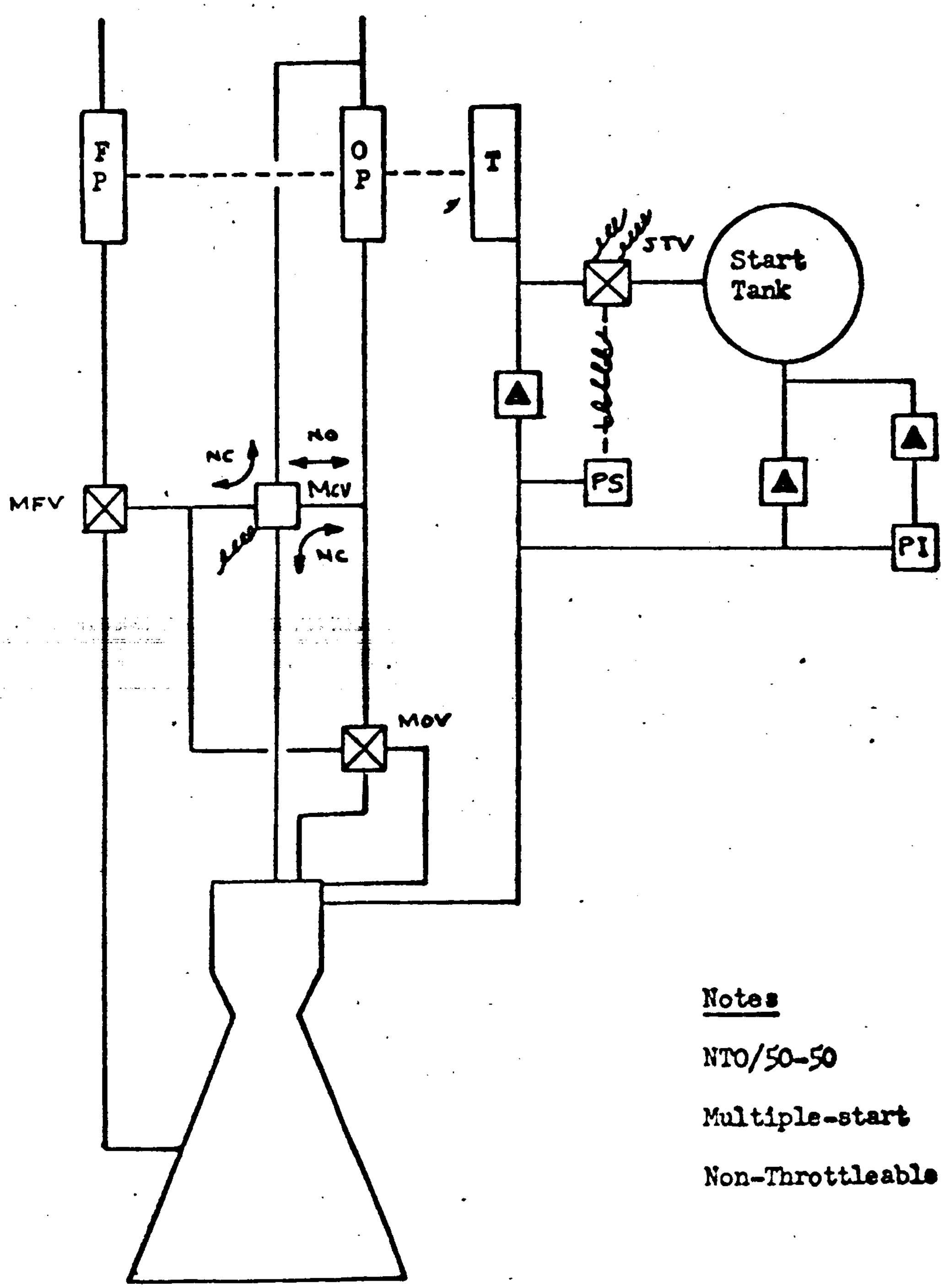
NTO/50-50

Multiple-start

Throttleable

Fig. 17

System 305



Notes

NTO/50-50

Multiple-start

Non-Throttleable

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Fig.

System 306

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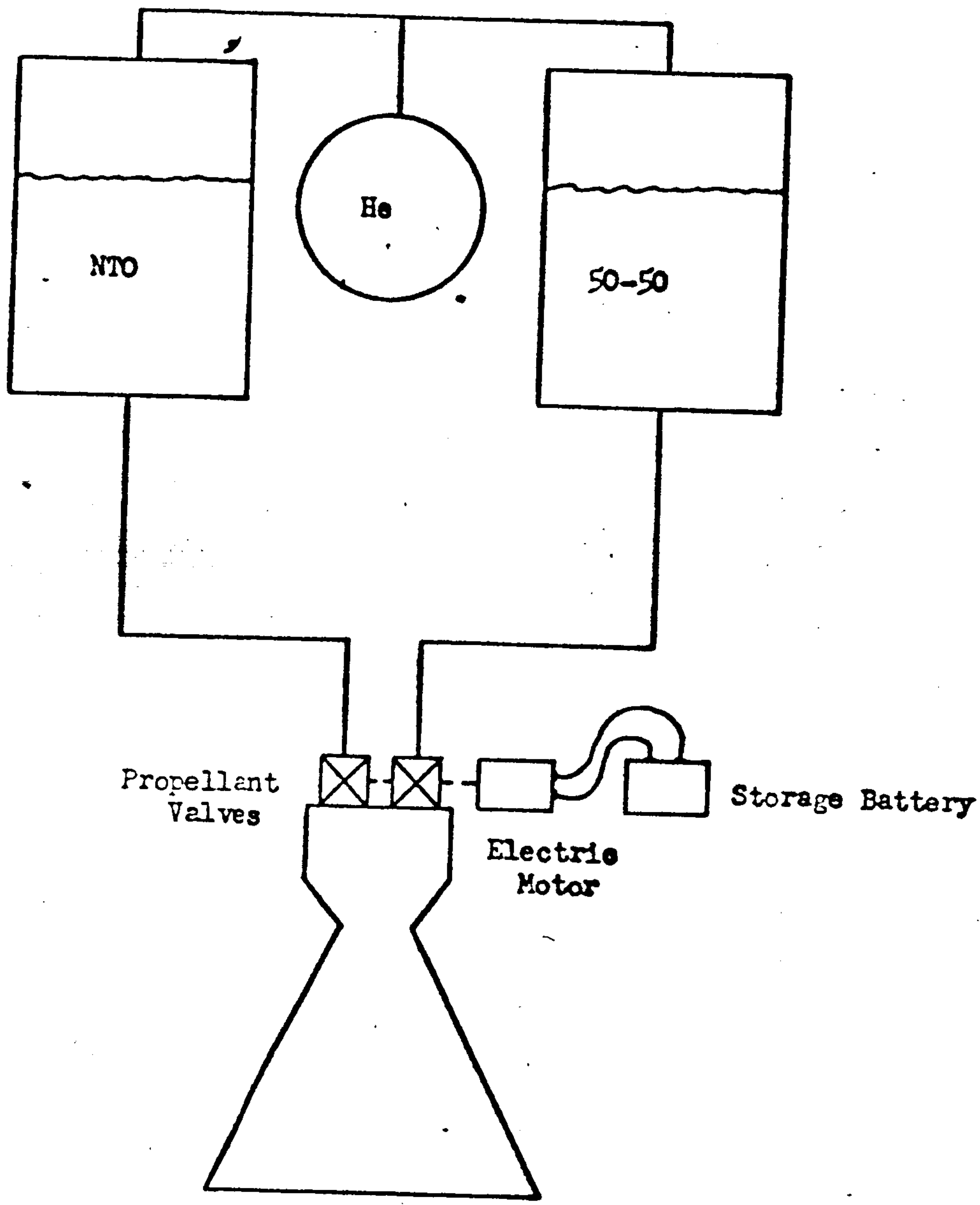

Notes

NTO/50-50

bootstrap; actuation of this switch (which is electrically linked to the STV) de-energizes (closes) the STV.

(8) The system bootstraps and mainstage is established.

System 302 differs from 301 in that a bi-propellant gas-generator start-system is used instead of a monopropellant system.

System 303 is identical to 302 except that the bi-propellant start-tanks are sized only for a single start, and are refilled during mainstage operation for the next start.

System 304 is identical to 301 except for the addition of a hot-gas throttling valve in the tap-off line.

System 305 differs from 101 in that it uses stored tap-off gases for turbine power during start. These gases are extracted from the tap-off line during engine operation; the pressure intensifier (PI) is used to increase the storage pressure above that available from the tap-off line.

System 306 is a pulsing-engine system. The pulsing engine is a pressure-fed, high chamber-pressure, low tank-pressure, pulsing engine (see section on concept evaluation). The propellant valves are sized so a "large" amount of propellant flows into the chamber before chamber pressure starts to rise. The propellant valves are closed, ignition occurs, and chamber pressure rises to a value much higher than the tank pressure.

Chamber pressure decays, the propellant valves open (admitting a new charge of propellant), and the cycle is repeated.

System 307 is identical to 305 except that it does not use a pressure intensifier in the start-tank recharging line.

System 401 is very much like the O_2/H_2 booster system, system 103. It differs in that igniter fuel is used instead of igniter oxidizer; i.e., the main oxidizer valve opens first, whereas in 103 the main fuel valve opens first.

RATING AND SELECTION

The basic systems defined in the preceding section have been rated (relative to one another) for operational simplicity, weight, and reliability. In addition a number of modified configurations (designated systems 101-A, 101-B, etc.) have been rated. These modifications were made to increase reliability, and are described in the footnotes to Table 2.

The purpose of this rating was to determine which systems were "optimum" on the basis of the above criteria. In defining overall ratings, the reliability ratings were weighted (as described below) to determine if this altered the ranking of the various systems.

Operational-simplicity ratings were defined on the basis of: (1) the number of major system components, (2) the number of events in the start

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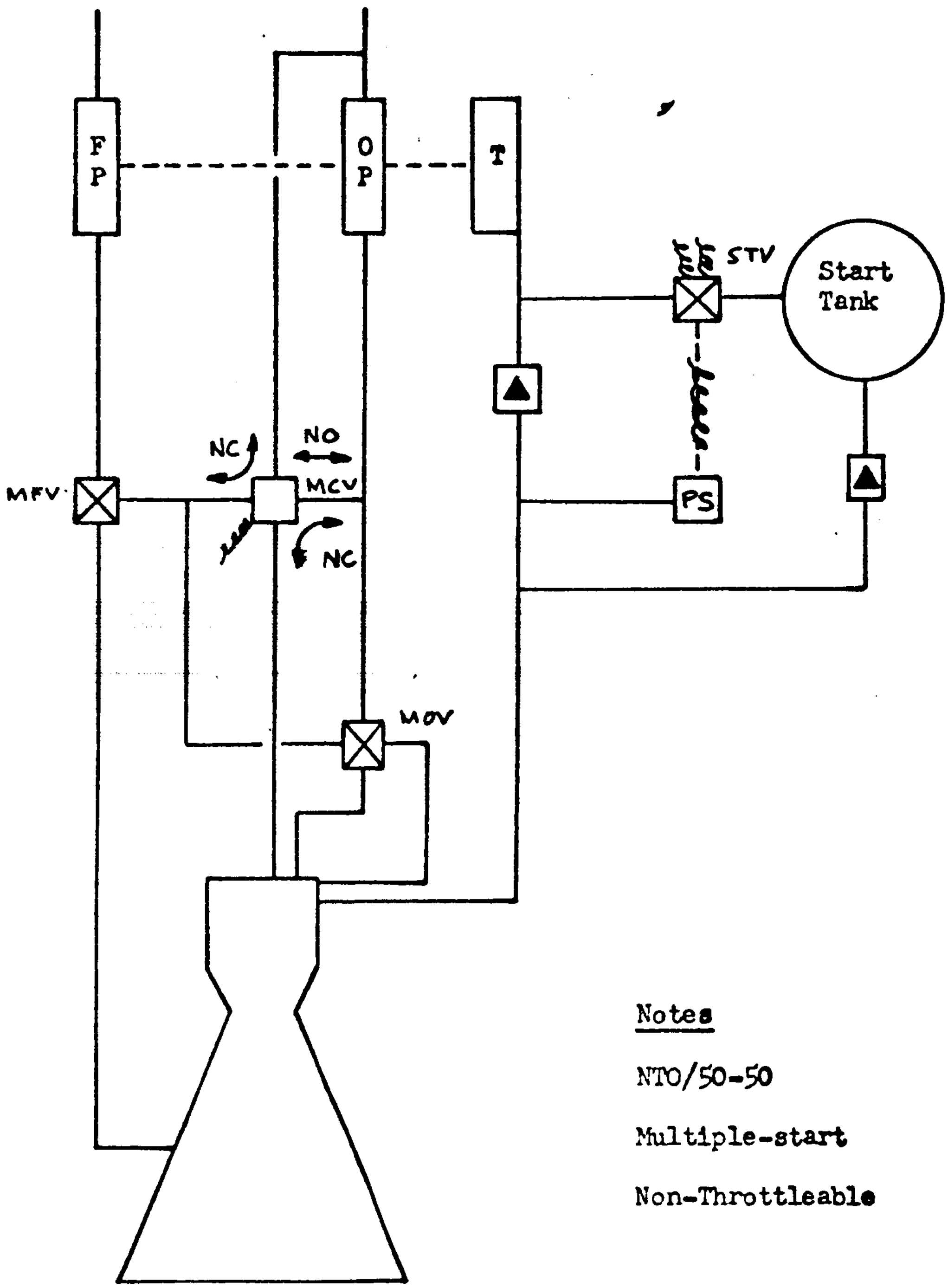

Notes

NTO/50-50

Multiple-start

Non-Throttleable

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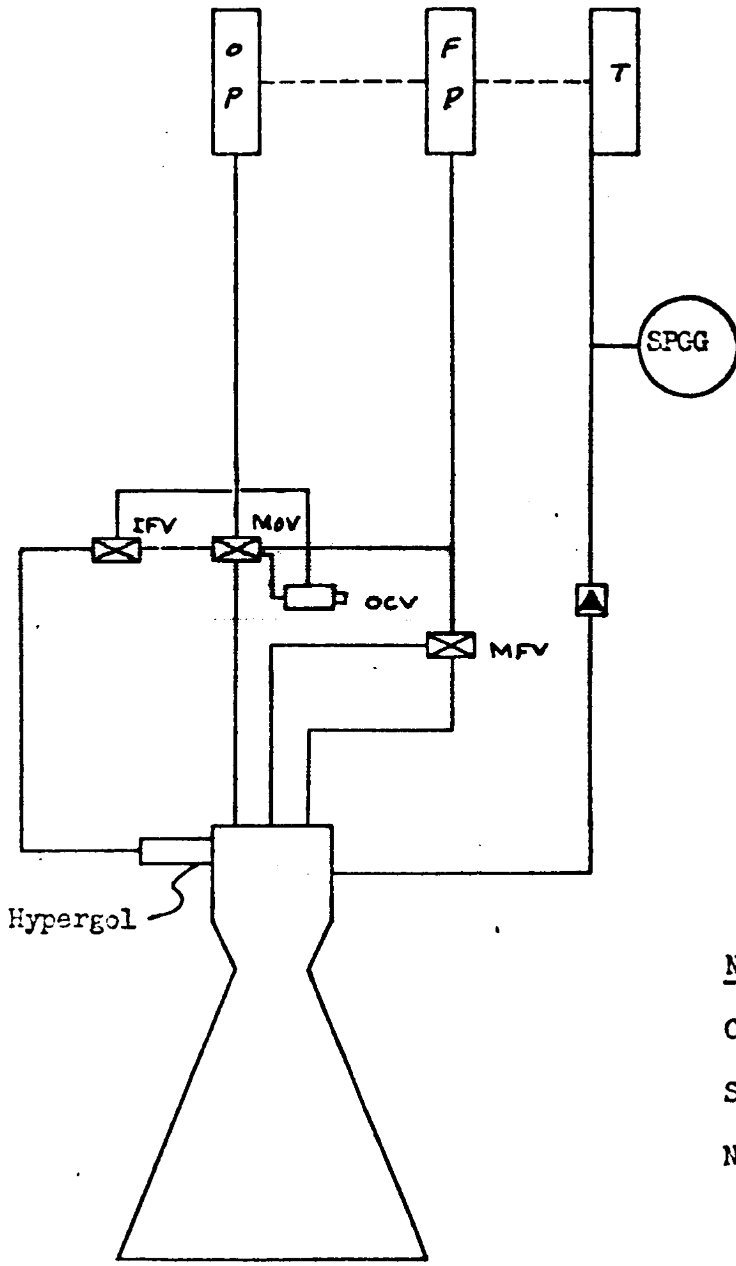

Notes

O_2/RP-1

Single-start

Non-throttleable

sequence, and (3) the number of events in the cutoff sequence (simpler systems have higher ratings).

The detailed rating method consists of assigning a separate percentile rating to each system in each category (operational simplicity, etc.). That is, all the systems are ranked (in each category), and a system's percentile rating indicates what percent of the total number of systems ranks below it. The unweighted over-all system rating is simply the average of the three (one for each category) percentile ratings. Weighted ratings are obtained by multiplying the three ratings by suitable factors, then computing the over-all rating.

For example, consider a system having unweighted ratings of 60, 70, 80 for operational simplicity, weight and reliability, respectively. Let us base the weighting on reliability being 1.5 times as important as operational simplicity, and low weight 2.0 times as important as operational simplicity. Unweighted and weighted ratings would be computed as shown below:

Unweighted (i.e., equal weighting)---

$$\text{Rating} = \frac{1(60) + 1(70) + 1(80)}{1 + 1 + 1} = 70$$

Weighted---

$$\text{Rating} = \frac{1(60) + 2(70) + 1.5(80)}{1 + 2 + 1.5} = 71.1$$

Ratings for the basic systems and modified systems are presented in Table 2 . The results of weighting reliability by a factor of 4.0, and operational simplicity and weight by a factor of 1.0 are also shown. This weighting was considered because it is anticipated that the desire for greater reliability is the factor likeliest to affect system selection for the types of systems considered in this program. In addition, it was found that the factor of 4.0 was the smallest value that would alter the system rankings based on over-all ratings.

As can be seen from the data in Table 2 , the variations in over-all ratings among any basic system and its modifications are small. For example, consider system 201 and its two modifications, 201-A and 201-B; the range is 68.57 to 71.35 for the unweighted and 59.31 to 64.87 for the weighted ratings.

Because the ratings are so similar for the various basic systems and their modifications, only basic systems have been selected for preliminary-design and layout. It is apparent from the nature of the modifications that integration concepts evolved for the basic configurations should be readily adaptable to the modified configurations, so no real limitation results from using the basic systems in this manner.

TABLE 2

SYSTEM RATINGS

System	Wt. Rating	Opor.-Simp. Rating	Reliability Rating	Props.	Multi-Start	Throttleable	Comments	*C	*D
101	66.72	33.36	88.96	O_2/H_2	Yes	No	Spacecraft	63.01	75.99
101-A	63.94	25.02	91.74	O_2/H_2	Yes	No	Spacecraft (1)	60.23	75.99
101-B	61.16	11.12	97.22	O_2/H_2	Yes	No	Spacecraft (1, 2, 3, 4)	56.50	76.86
102	38.92	25.02	66.72	O_2/H_2	Yes	Yes	Spacecraft	43.55	55.14
102-A	36.14	11.12	69.50	O_2/H_2	Yes	Yes	Spacecraft (1)	38.92	54.21
102-B	33.36	0.00	77.84	O_2/H_2	Yes	Yes	Spacecraft (1, 2, 3, 4)	37.07	57.45
103	80.62	91.74	5.56	O_2/H_2	No	No	Booster	59.27	32.43
103-A	77.84	86.18	8.34	O_2/H_2	No	No	Booster (1)	57.45	32.90
104	58.38	25.02	86.18	O_2/H_2	Yes	No	Spacecraft	56.53	71.35
105	52.82	11.12	83.40	O_2/H_2	Yes	No	Spacecraft	49.78	66.26

TABLE 2 (Cont'd)

System	Wt. Rating	Oper.-Simp. Rating	Reliability Rating	Props.	Multi-Start	Throttleable	Comments	*C	*D
201	75.06	91.74	47.26	F_2/H_2	Yes	No	Spacecraft	71.35	59.31
201-A	72.28	86.18	52.82	F_2/H_2	Yes	No	Spacecraft (1)	70.43	61.62
201-B	69.50	75.06	61.16	F_2/H_2	Yes	No	Spacecraft (1, 2, 3, 4)	68.57	64.87
202	47.26	75.06	13.90	F_2/H_2	Yes	Yes	Spacecraft	45.41	29.65
202-A	44.18	61.16	16.68	F_2/H_2	Yes	Yes	Spacecraft (1)	40.77	28.68
202-B	41.70	52.82	19.46	F_2/H_2	Yes	Yes	Spacecraft (1, 2, 3, 4)	37.99	28.73
203	55.60	86.18	25.02	F_2/H_2	Yes	No	Spacecraft	55.60	40.31
204	50.04	75.06	22.24	F_2/H_2	Yes	No	Spacecraft	49.11	35.68
205	94.52	94.52	72.28	F_2/H_2	Yes	No	Spacecraft	87.11	79.69

TABLE 2 (Cont'd)

System	Wt. Rating	Oper.-Simp. Rating	Reliability Rating	Props.	Multi-Start	Throttleable	Comments	*C	*D
301	30.58	61.16	0.00	NTO/ 50-50	Yes	No	Spacecraft	30.58	15.29
301-A	27.80	52.82	2.78	NTO/ 50-50	Yes	No	Spacecraft (1)	27.80	15.29
301-B	25.02	33.36	11.12	NTO/ 50-50	Yes	No	Spacecraft (1, 2, 3, 5)	23.17	17.14
302	22.24	52.82	50.04	NTO/ 50-50	Yes	No	Spacecraft	41.70	45.87
302-A	19.46	41.70	55.60	NTO/ 50-50	Yes	No	Spacecraft (1)	38.92	47.26
302-B	16.69	25.02	63.94	NTO/ 50-50	Yes	No	Spacecraft (1, 2, 3, 5)	35.21	49.58
303	11.12	41.70	36.14	NTO/ 50-50	Yes	No	Spacecraft	29.65	32.90
303-A	5.56	33.36	41.70	NTO/ 50-50	Yes	No	Spacecraft (1)	26.87	34.29
303-B	0.00	11.12	58.38	NTO/ 50-50	Yes	No	Spacecraft (1, 2, 3, 5, 6)	23.17	40.77

TABLE 2 (Cont'd)

System	Wt. Rating	Oper.-Simp. Rating	Reliability Rating	Props.	Multi-Start	Throttleable	Comments	*C	*D
304	13.90	52.82	30.58	NTO/ 50-50	Yes	Yes	Spacecraft	32.43	31.51
304-A	8.34	41.70	38.92	NTO/ 50-50	Yes	Yes	Spacecraft (1)	29.65	34.29
304-B	2.78	25.02	44.18	NTO/ 50-50	Yes	Yes	Spacecraft (1, 2, 3, 4)	23.99	34.09
305	83.40	61.16	77.84	NTO/ 50-50	Yes	No	Spacecraft	74.13	75.99
306	97.22	97.22	94.52	NTO/ 50-50	Yes	No	Spacecraft	96.33	95.42
307	86.18	75.06	80.62	NTO/ 50-50	Yes	No	Spacecraft	80.62	80.62
401	91.75	86.18	27.80	O_2/ RP-1	No	No	Booster	68.57	48.19
401-A	88.96	75.06	36.14	O_2/ RP-1	No	No	Booster (2)	66.72	51.43

*C - Over-all rating with factors equally weighed.
*D - Over-all rating with reliability weighed 4 and other factors weighed 1.

1. System includes redundant pressure switches.
2. System includes series - parallel check valves in tap-off line.
3. System includes double solenoids in control valves.
4. System includes redundant check valves in start-tank refill line.
5. System includes series - parallel check valves from expulsion tank to gas generator.
6. System includes series - parallel check valves in expulsion tank refill lines.

CONCEPT INVESTIGATION AND EVALUATION

Analytical and design investigations and evaluations have been made for a number of concepts including those selected in the preceding section.

Analytical investigations generally consisted of determining feasibility and performance, and of making payload comparisons with more conventional systems.

Design investigations consisted primarily of investigating means of physically integrating components and subsystems for the more promising basic systems.

NOMINAL PARAMETER SELECTION

Nominal values of thrust, chamber pressure, and mixture ratio were selected to provide a basis for sizing various components during concept evolution. Two sets of nominal values of thrust, chamber pressure, area ratio and mixture ratio were selected; one for spacecraft, the other for boosters. The selected values are listed in Table 3 and are discussed below in some detail.

TABLE 3

SPACECRAFT (THRUST = 40K)

Propellants	Chamber Pressure	Area Ratio	Mixture Ratio
LO_2/LH_2	500	160	5.5
LF_2/LH_2	500	180	10.0
N_2O_4/N_2H_4-UDMH(50-50)	500	165	2.1
BOOSTERS (THRUST = 6M)			
LO_2/LH_2	1800	45	5.0
LO_2/RP-1	2200	55	2.24

The thrust-level of 40K for spacecraft was selected because it is a value frequently mentioned as appropriate for a high-energy upper-stage engine. Furthermore, this is a thrust-level which is reasonably representative of both the smallest pump-fed spacecraft engines being considered today, and the largest spacecraft engines that can be used with presently planned lower stages.

The chamber pressure of 500 was chosen because it is sufficiently high to show a distinct size advantage over optimum pressure-fed systems and high enough to give near-optimum payloads* (within one percent of optimum,

*Based on ΔV = 10,000 fps and F/W = 0.4

Ref. 1). It should be noted that this chamber pressure may have to be reduced for some nozzle configurations if regeneratively-cooled chambers are used.

Area ratios were selected to be consistent with a payload that was one percent less than optimum (Ref. 1).

Mixture ratios were selected using the method of Ref. 1 wherein a compromise value between the optima for frozen performance and equilibrium performance is chosen so as to minimize the potential payload loss.

Basing the above selections on a payload that is one percent less than optimum, was done somewhat arbitrarily. However, it is felt that this is justifiable, since the parameters are being used primarily for sizing components, etc.

Booster parameters were selected on a similar basis. The chamber pressures are a little lower than optimum values, in the interest of reliability; these values were chosen such that the required coolant pressure drop was approximately equal to the chamber pressure.

NOMINAL FLOWRATES

Nominal system flowrates were determined for a representative group of basic systems to provide data for use in sizing individual components.

These flowrates are based on mass and energy balances for steady-state operation of the listed systems.

Engine performance and flows, etc., are presented in Table 4 below:

TABLE 4

ENGINE PERFORMANCE

System	203	302	103	401
Propellants	LF_2/LH_2	NTO/50-50	LO_2/LH_2	LO_2/RP-1
I_e	437.7	338.5	380.7	295.9
F	40K(VAC)	40K(VAC)	6M(SL)	6M(SL)
Nozzle Type	Bell	Bell	Aerospike	Aerospike
P_c	500	500	1800	2200
ϵ	180	165	45	55
MR_e	10.0	2.1	5.0	2.24
$\dot{W}_{oe}$	83.1	38.1	13,134	14,020
$\dot{W}_{fe}$	8.3	80.1	2627	6259
$\dot{W}_t$	91.4	118.1	15,761	20,279

where:

- I_e - engine specific-impulse
- F - engine thrust, vacuum (VAC) or sea level (SL)
- P_c - chamber pressure (psia)
- ϵ - nozzle expansion-area ratio
- MR_e - engine mixture-ratio (oxidizer/fuel)
- $\dot{w}_{oe}$ - total engine oxidizer-flowrate (lbs/sec)
- $\dot{w}_{fe}$ - total engine fuel-flowrate
- $\dot{w}_t$ - turbine(s) flowrate

ANALYTICAL INVESTIGATIONS

Start Systems

A start-system analysis was made to estimate relative start-times, weights, and volumes for a number of possible multiple-start gas-spin start-systems, and to get an indication of the feasibility of the novel start system designated tap-off gas spin-start. This system (Figure 21) could provide an extremely simple means of restarting an NTO/50-50 spacecraft system. This system would extract fuel-rich gases from the thrust chamber and store them in a pressure bottle for use in spinning the turbine for the next start. The results of the analysis served to guide the more detailed definition of the selected concepts. The propellant combinations considered were LO_2/LH_2, LF_2/LH_2 and N_2O_4/N_2H_4-UDMH(50-50). The systems were rated on the basis of time required to start (defined as 90% of rated thrust),

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Fig. 21

Tap-off Gas Spin-Start System

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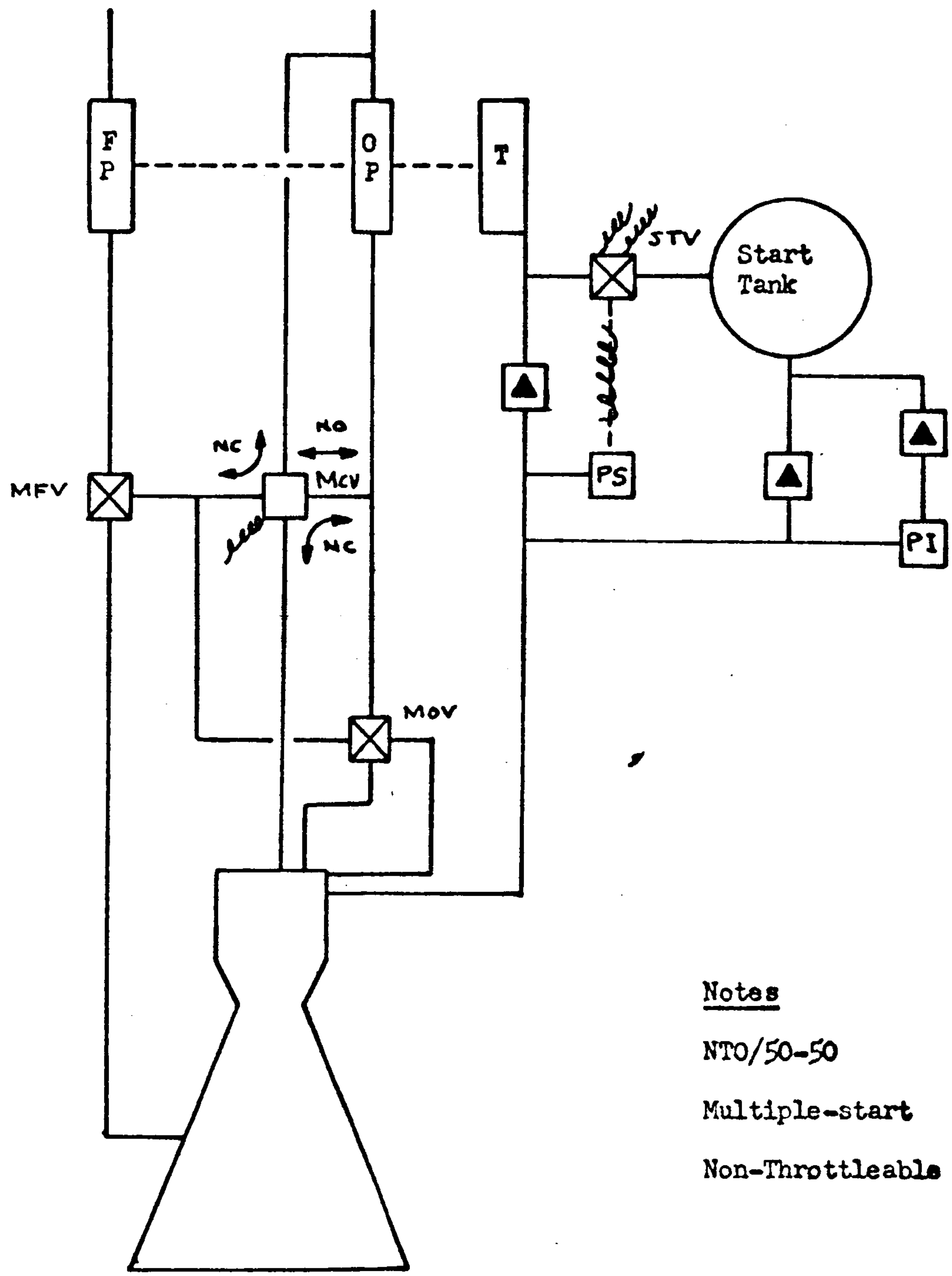

Notes

NTO/50-50

Multiple-start

Non-Throttleable

weight and volume.

For LO_2/LH_2 and LF_2/LH_2 systems, stored gaseous-hydrogen at 3000 psia and $400^{\circ}R$ regulated to 600 psia during start-up is rated highest. The high storage-pressure is achieved using a reciprocating-piston pressure intensifier to increase the pressure above that obtainable directly from the thrust-chamber cooling-jacket. This system is shown achematically in Fig. 22 .

For the NTO/50-50 system, start-tank blowdowns from pressures of 1000 psia, 800 psia, and 400 psia (all at $800^{\circ}R$) were all rated equally. Since these data are based on a system with a chamber pressure of 500 psia, the first systems would require pressure intensifiers as shown in Figure 21 . Therefore, blowdown from 400 psia would be "optimum" based on operational simplicity, weight, and reliability.

Table 5 contains a list of the systems analyzed:

Table 5

Propellants	Pressurant	Temperature ($^{\circ}R$)	Control
F_2/H_2	H_2	400	Regulated
	H_2	400	Blowdown
	F_2/H_2	1460	Blowdown
O_2/H_2	H_2	400	Regulated
	H_2	400	Blowdown
	O_2/H_2	1460	Blowdown
NTO/50-50	NTO/50-50	800	Blowdown

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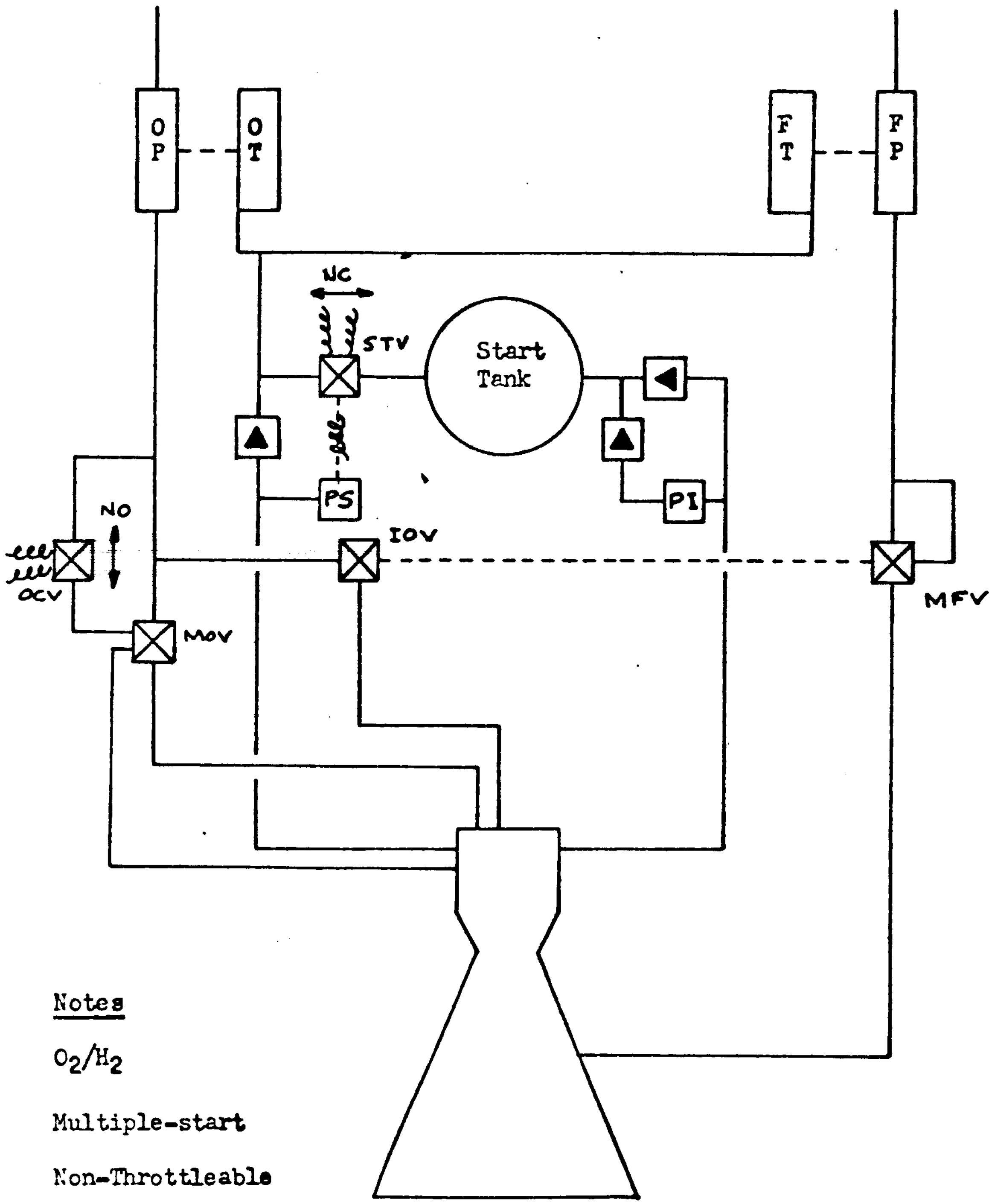
DATE	Jacket-Gas Gas-Spin Start System	MODEL NO



Notes

O_2/H_2

Multiple-start

Non-Throttleable

For each regulated system, storage pressures of 1200, 1800, 2400, and 3000 psia were considered. The bottle storage pressure was regulated to 600 psia. For each blowdown system, storage pressures of 600 and 1200 psia were considered with storage volumes of 2000 and 4000 cubic inches each.

Start times, weights, volumes, and percentile ratings for the systems are presented in Table 6 . It is of interest to note that the hydrogen system in spite of the lower temperature than either bipropellant system (LO_2/LH_2 and LF_2/LH_2) provides a more rapid start due to its superior molecular weight and specific heat.

The analysis for the tap-off gas spin-start system described (in part) above indicated that a 40K engine could probably be started using tap-off gases initially stored at a temperature of approximately 400°F with the re-start occurring before the temperature drops below 70°F. It may be that the gases could be stored at an even higher initial temperature, which would be desirable, since storage at 400°F could require cooling the gases before storage. This was not investigated during this study. However, further work is required (see Vol. I -- Suggested Additional Work) to substantiate the results of these analyses. The primary uncertainties involved are tap-off gas properties for the conditions described.

TABLE 6

Description				Time		Volume		Weight		Over-all Rating
Propellants	Process	Gas	Press.	Sec.	*Rating	In^3	Rating	Lb	Rating	
F_2/H_2	Reg (400°R)	H_2	1200	1.6	50	3700	33	69	0	28
			1800		50	1850	75	57	8	44
			2400		50	1240	83	54	17	50
			3000		50	925	92	53	25	56
	Blowdown (1460°R)	F_2/H_2	600	2.80	0	2000	42	16	83	42
				2.25	17	4000	0	32	50	22
	Blowdown (1460°R)	F_2/H_2	1200	2.05	33	2000	42	26	66	47
				1.6	83	4000	0	42	33	38
	Blowdown (400°R)	H_2	600	2.65	8	2000	42	16	83	44
				2.05	17	4000	0	32	58	25
	Blowdown (400°R)	H_2	1200	1.9	42	2000	42	26	66	50
				1.45	92	4000	0	42	33	42

*Percentile rating done independently for each propellant combination.

TABLE 6 (Cont'd)

Description				Time		Volume		Weight		Over-All Rating
Propellants	Process	Gas	Press.	Sec.	Rating	In^3	Rating	Lb	Rating	
O_2/H_2	Reg (400^oR)	H_2	1200	1.6	58	3700	33	69	0	30
			1800		58	1850	75	57	8	47
			2400		58	1240	83	54	17	53
			3000		58	925	92	53	25	58
	Blowdown (1460^oR)	O_2/H_2	600	2.9	0	2000	42	16	83	42
				2.3	8	4000	0	32	50	19
	Blowdown (1460^oR)	O_2/H_2	1200	2.1	25	2000	42	26	66	44
				1.65	50	4000	0	42	33	28
	Blowdown (400^oR)	H_2	600	2.65	17	2000	42	16	83	47
				2.05	33	4000	0	32	58	30
	Blowdown (400^oR)	H_2	1200	1.9	42	2000	42	26	66	50
				1.45	92	4000	0	42	33	42

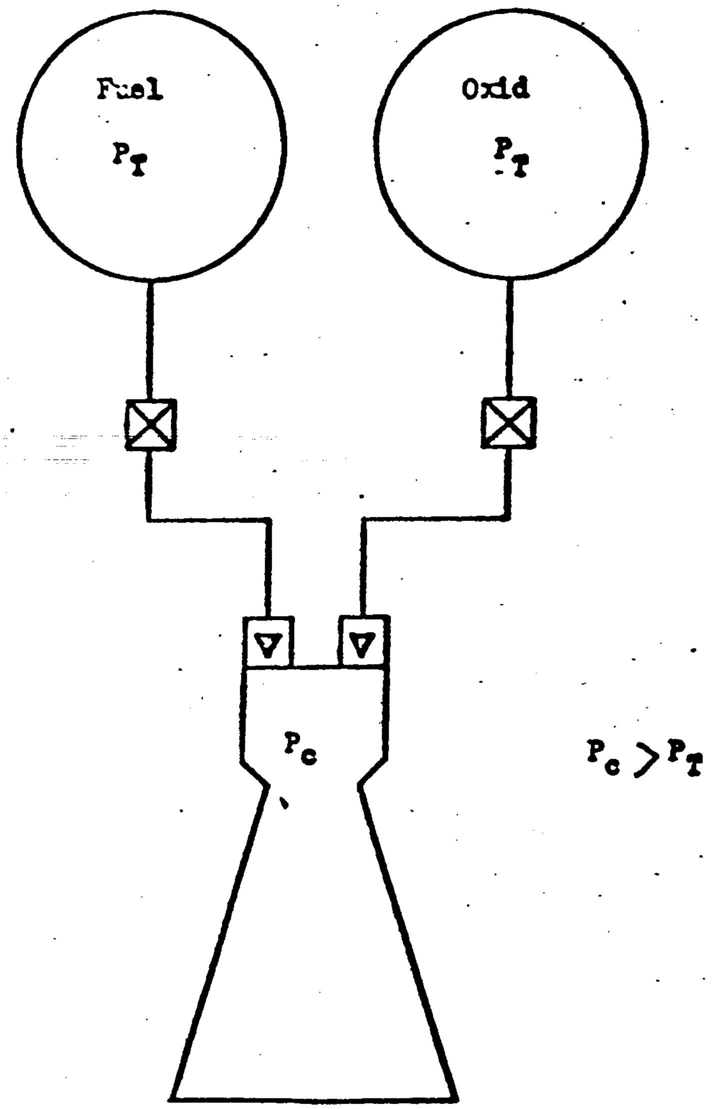
Pulsing Engine

The pulsing engine is a pressure-fed, high chamber-pressure, low tank-pressure, pulsing engine. The primary features of the engine are check valves near the injector and main propellant-valves (Fig. 23). The check valves are sized so a "large" amount of propellant flows into the chamber before chamber pressure builds up enough to close the check valves. The propellant burns and the chamber pressure builds up to a large value (considerably greater than tank pressure) which closes the check valves. Chamber pressure decays, the check valves open, and the cycle is repeated. This configuration probably represents the simplest form this concept could assume. A slightly less simple configuration, Figure 24 should be easier to test and develop. This configuration operates in a similar fashion; the only difference being that propellant flow is regulated by an electrically actuated valve. Evaluation of the pulsing-engine concept was based on the latter configuration (Figure 24), and consisted estimating performance and operating characteristics for a range of thrusts.

This analysis was conducted using a modified start-transient computer program with distributed-parameter flow equations. All systems considered had total propellant flow areas that were equal to the thrust-chamber throat area.

Figure 23

Schematic-- Pulsing Engine



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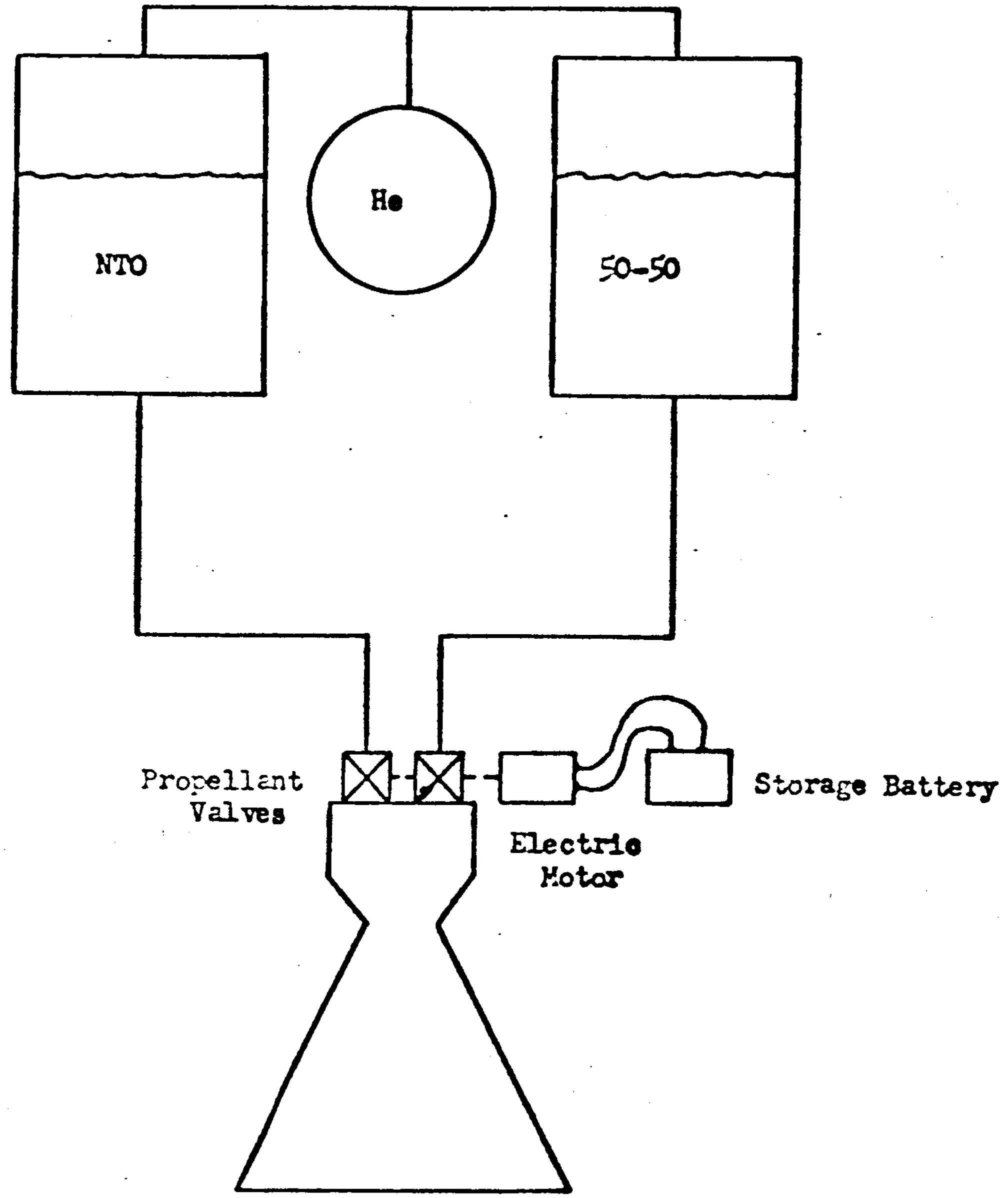
DATE:	Schematic-Pulsing Engine	MODEL NO.



Notes

NTO/50-50

The results of this analysis are summarized in Figure 25 . These data also indicated that the time-average (effective) specific-impulse (for the part of the cycle in which thrust is being produced) was 95 to 98% of the maximum specific impulse obtained during the cycle. This assumes that combustion at each transient condition is comparable to what would be obtained for steady-state operation under the same conditions. Four basic systems were analyzed using the NTO/50-50 propellant combination. These systems had throat areas of 0.1, 1.0, 10.0, and 100.0 in^2. From the family of curves in Figure 25, it is possible to determine the combustion cycle-rate and/or ignition delay required to obtain a given maximum chamber pressure or maximum thrust. The time-average thrusts are also plotted. Trends in all these system parameters can be readily identified from this plot.

The effect of combustion-chamber geometry was investigated by varying the combustion-chamber characteristic length (L^*) for one of the basic systems. The effects of this variation on maximum chamber pressure, maximum thrust, and cycle rate are shown in Figure 26.

The effect of variations in propellant-line size was also investigated. It was found that, if all other system parameters were held constant, chamber pressure and thrust were directly proportional to total propellant line area. The cycle rate is reduced slightly as the line area increases due to a higher chamber pressure at the initiation of nozzle blowdown.

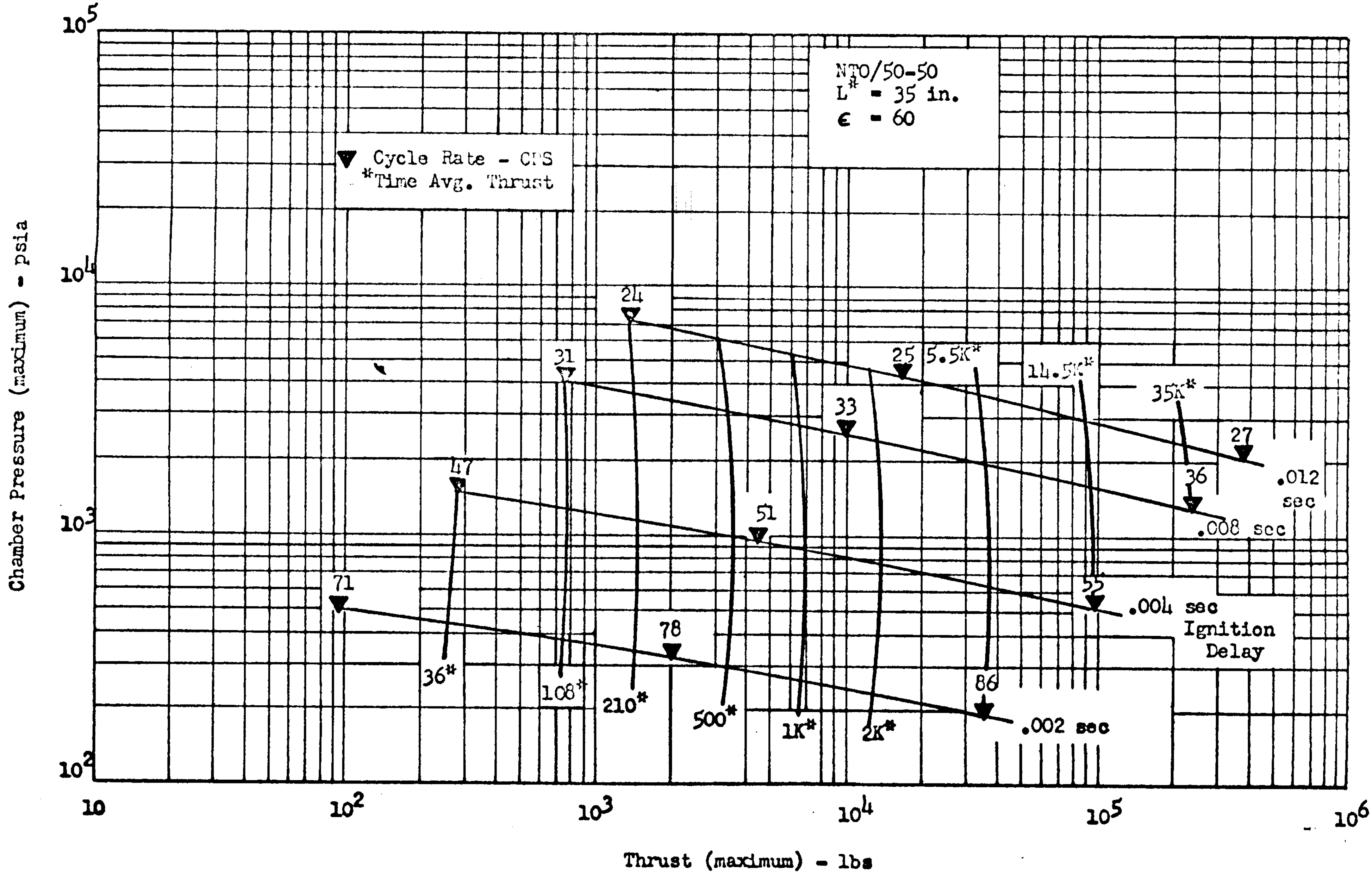


Figure 25. Pulsing-Engine Characteristics, Summary Plot

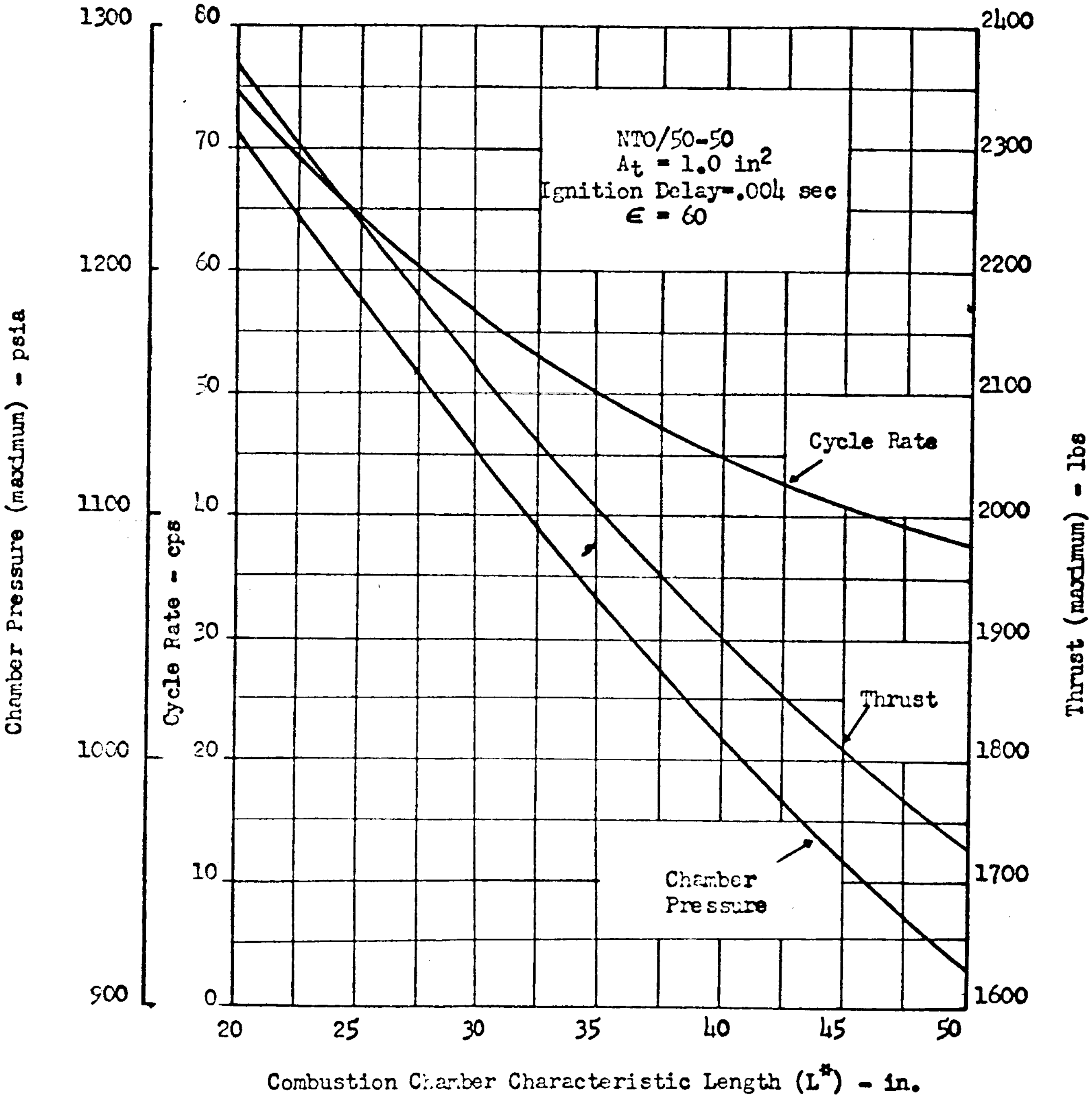


Fig. 26. Effects of Pulsing-Engine Chamber Geometry

For the system analyzed (A_T = 1.0 in^2 ignition delay = .004 sec), doubling the total propellant line area, reduced in cycle rate from 50 to 45 cps while decreasing the total propellant line area by a factor of two increased cycle rate from 50 to 55 cps.

Figure 27 presents the results of payload comparisons with optimum pressure-fed systems at thrust levels of .5K, 2K, and 40K pounds. System weights and operating parameters were obtained from the "Engine Operating Parameter Optimization Program" (Ref. 1). As is shown, the payload advantage increases with velocity increment and decreasing thrust.

For the most part, the payload advantage over pressure-fed systems is not appreciable. However, the reduction in propulsion system size is substantial (Figures 28 and 29).

The remaining figures in this section contain parametric data used to generate the summary curves (Figure 25), and are discussed briefly below.

Figure 30 presents a plot of maximum chamber pressure versus throat area for parametric ignition delay. As can be seen the maximum chamber pressure is a strong function of ignition delay.

Figure 31 presents a plot of maximum thrust versus throat area for parametric ignition delay. A thrust of nearly 400,000 pounds is obtained with a system which has a 100 in^2 throat area and a .012 second ignition delay.

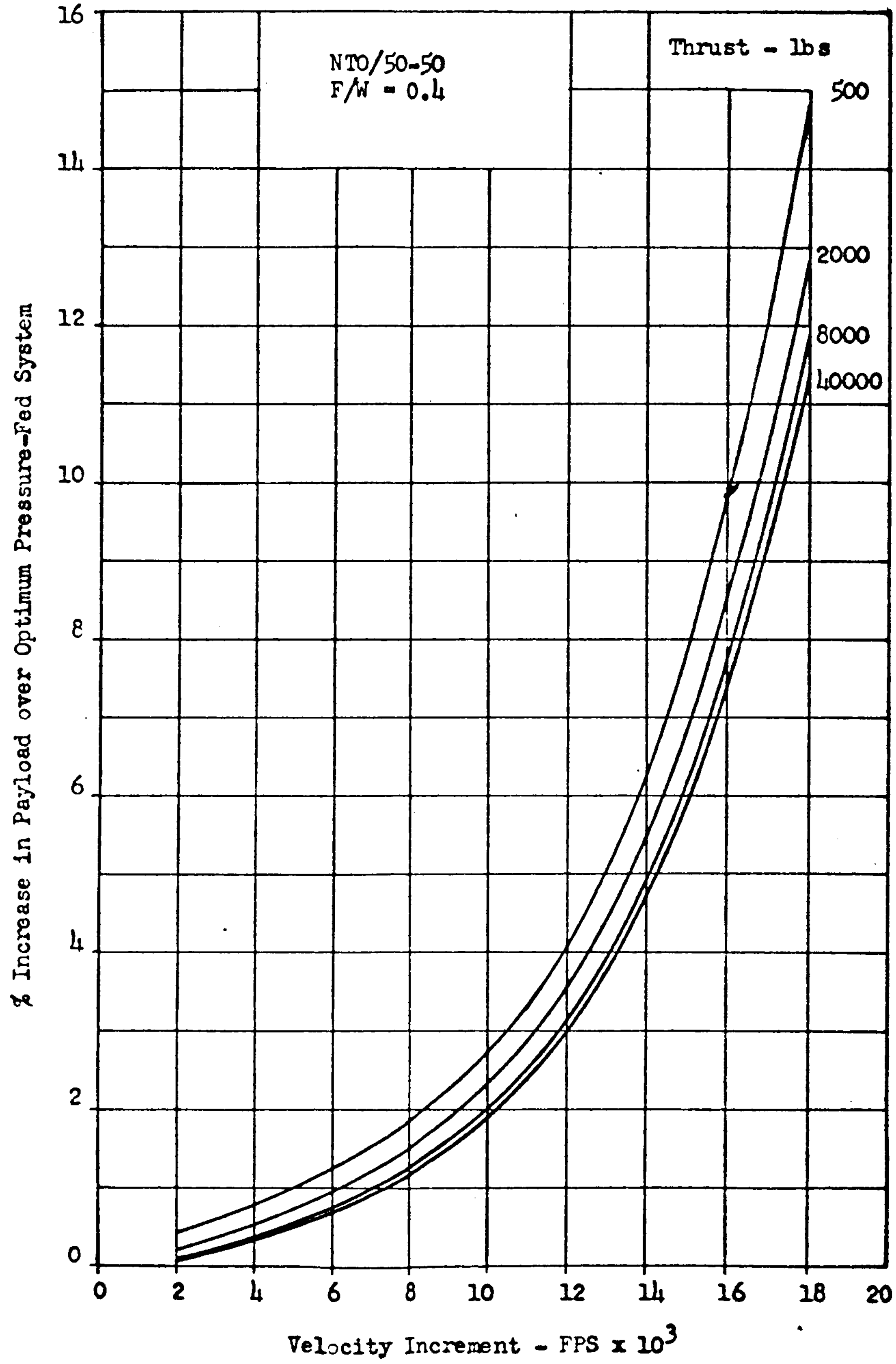


Fig. 27. Payload Comparison - - Pressure-Fed and Pulsing Engines

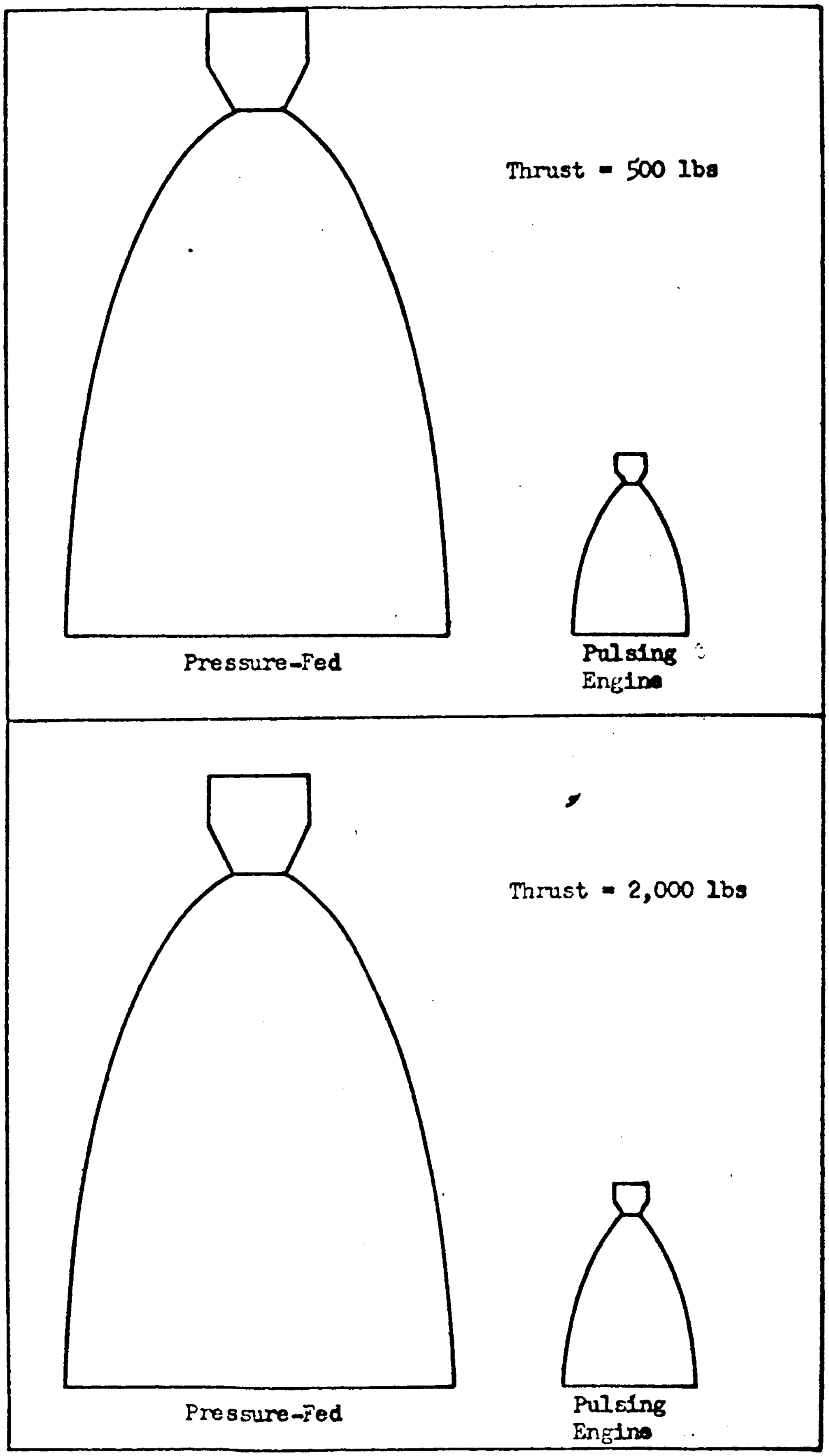


Fig. 28. Size Comparison - - Pressure-Fed and Pulsing Engines

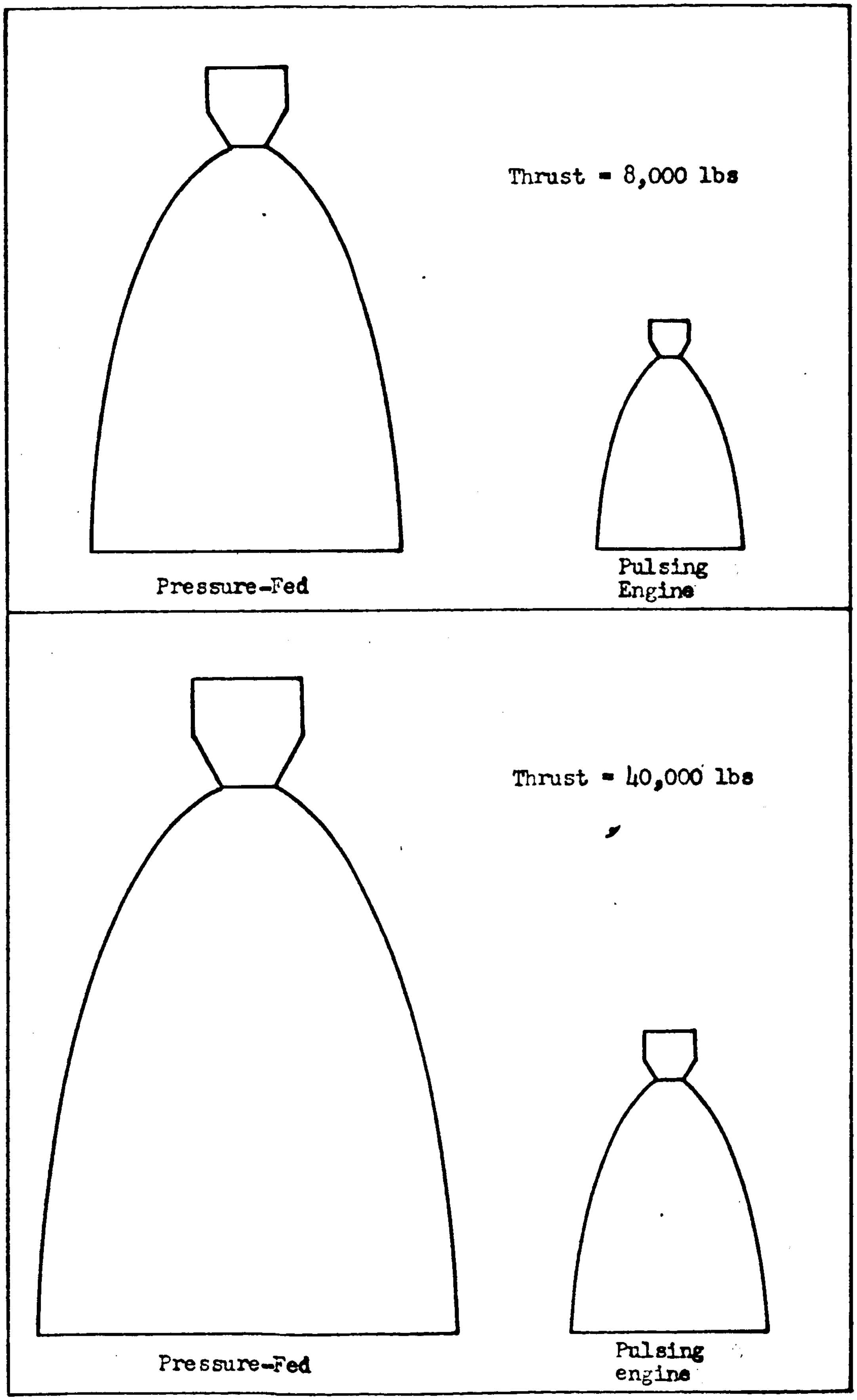


Fig. 29. Size Comparison - - Pressure-Fed and Pulsing Engines

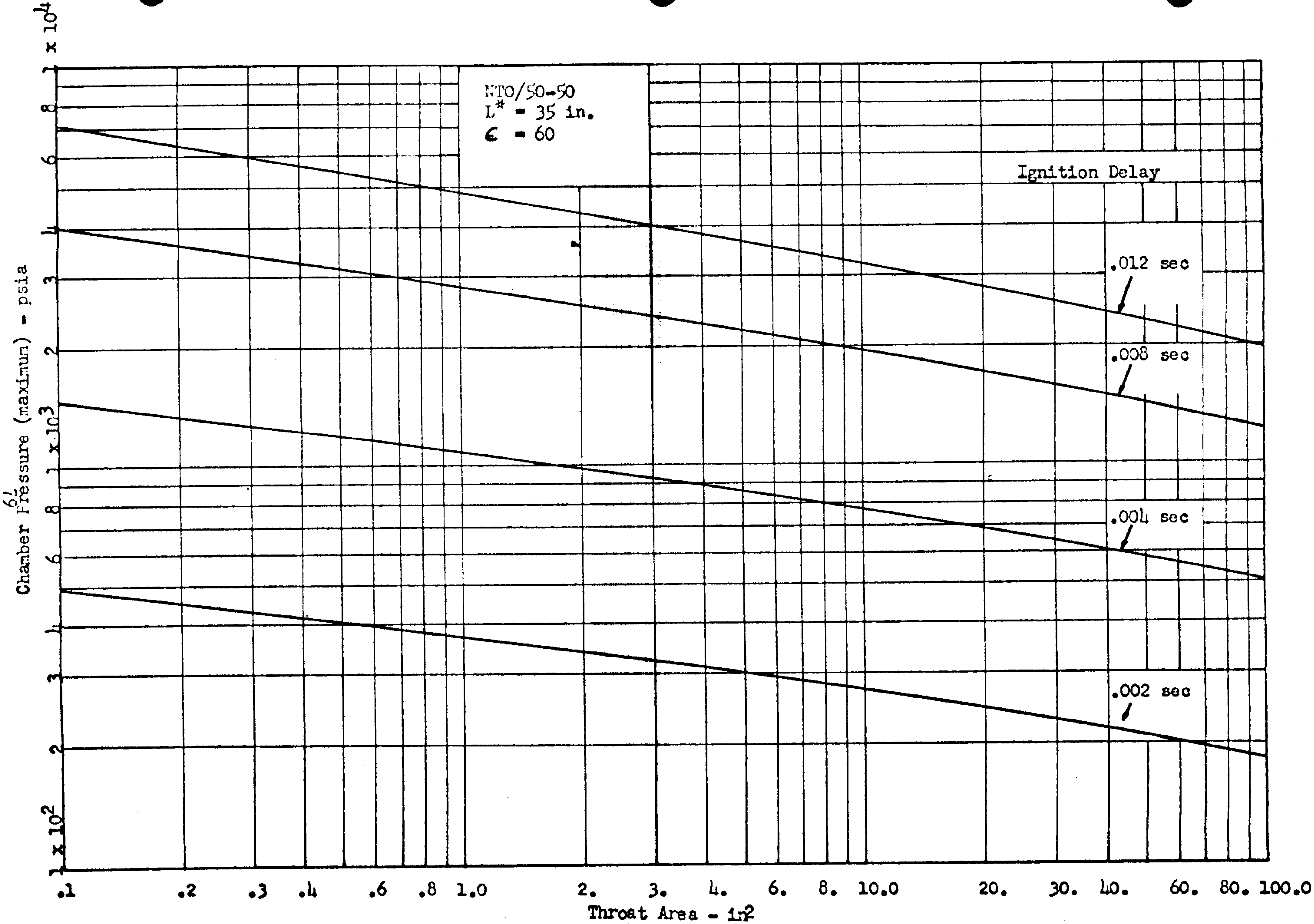


Fig. 30. Pulsing Engine, Chamber Pressure vs Throat Area

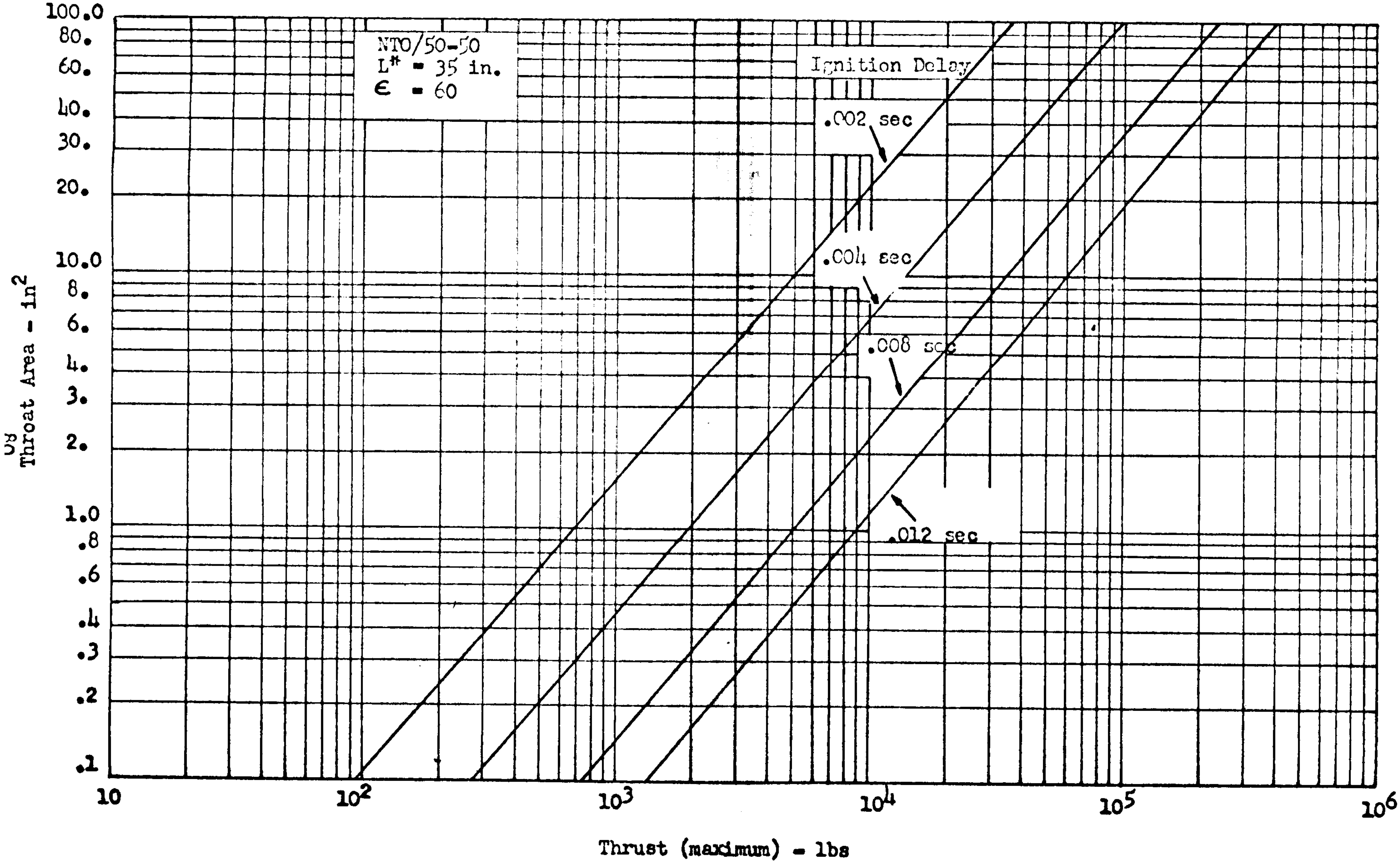


Fig. 31. Pulsing Engine, Thrust vs Throat Area

Figure 32 presents a plot of combustion cycle rate versus throat area for parametric ignition delay. Cycle rate is a strong function of the ignition delay.

Figure 33 presents a plot of time average thrust versus throat area. The chamber pressure required to produce the time average thrust (with the same throat area and area ratio) in a non-pulsing engine is also shown.

Pressure-Fed/Pump-Fed System

The combined pump-fed/pressure-fed system has a potential size and weight advantage over the all pressure-fed system, and a reliability advantage over the all pump-fed system. The advantage over pressure-fed systems results from the combination system having a higher optimum chamber pressure. Favorable comparison with pump-fed systems results from eliminating the unreliability associated with one of the turbopumps. This latter advantage appeared to be especially attractive for F_2/H_2 systems wherein the fluorine pump could be eliminated. This could significantly reduce the time and cost required to develope a high chamber-pressure (i.e., higher than pressure-fed optimum) F_2/H_2 system, since the current "state-of-the-art" of hydrogen pumps is fairly well advanced. For these reasons, a mission analysis and chamber-pressure area-ratio optimization were made to compare a pressure-fed and combined system using F_2/H_2. This optimization was made using the methods of Ref. 1.

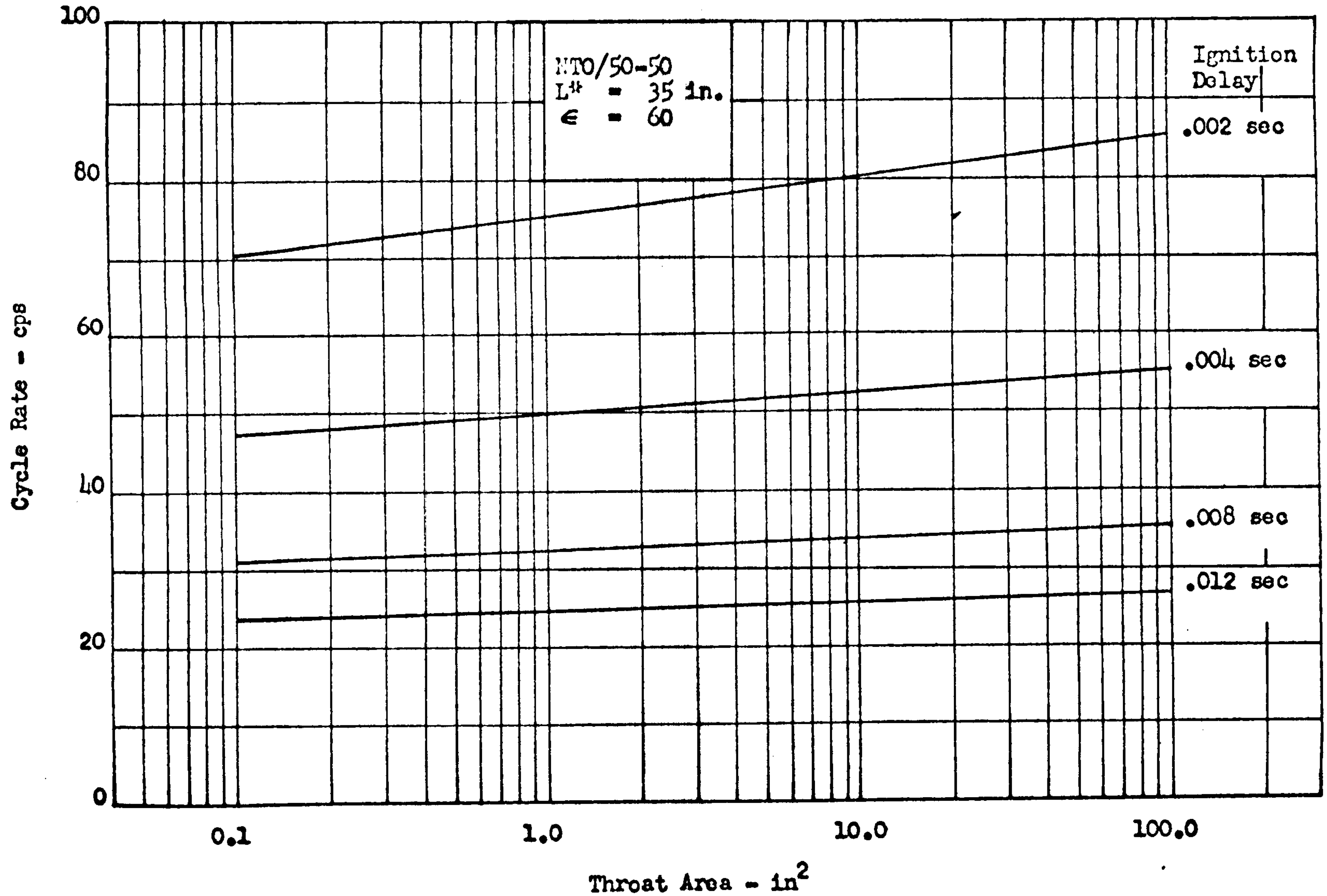


Fig. 32. Pulsing Engine, Cycle Rate vs Throat Area

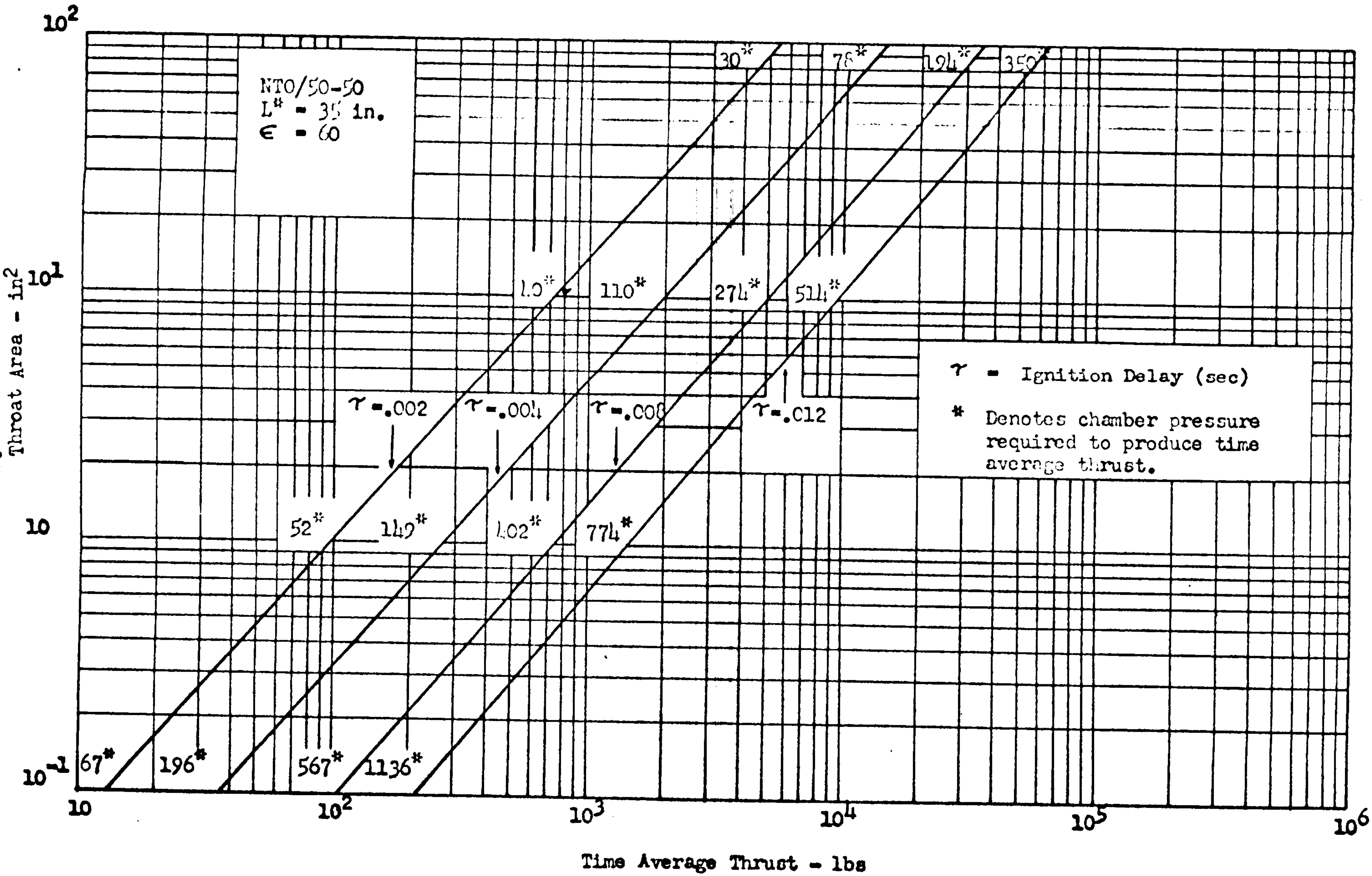


Fig. 33. Pulsing Engine, Average Thrust vs Throat Area

The results of the optimization portion of this analysis are presented in Table 7 below. These data are based on a thrust level of 40K, a stage initial thrust-to-weight ratio of 0.4, and a gross-velocity increment of 10,000 feet per second.

Table 7

Optimum Parameters	Pressure-Fed System	Combined System
Chamber Pressure	90	120
Area Ratio	120	140

The effect of going to the combined system (instead of a pressure-fed system) is as expected, chamber pressure and area ratio optimize at higher values. This effect could be utilized in either of two ways. Both the higher chamber pressure and the higher area ratio could be used, thereby improving performance and payload --- this payload increase is approximately 2.5% for the thrusts that were considered, viz., 40K and 60K. Alternatively, the higher chamber pressure and some area ratio smaller than optimum could be used to provide a propulsion system having a significantly smaller volume; this system would probably still possess a performance and payload advantage.

The attractiveness of this concept is, of course, directly linked to the attractiveness of the payload gain and/or size reduction associated with it. Under some circumstances, these gains may be sufficient to warrant utilization of the concept.

In a similar comparison with an optimum all pump-fed system, the combined system had an approximately 4% lower payload. Thus, in considering the use of the combined system instead of a pump-fed system, one must establish a tradeoff between reliability and payload.

A possible system configuration for this system is presented in the Design Section of this report.

Gas-Drive Jet-Pump

The gas-drive jet pump was investigated as a possible means of obtaining higher chamber pressures using a simple pumping device. It is, in essence, a device which converts the internal energy of a pressurized gas to kinetic energy of a liquid-gas stream, from which momentum can then be transferred to the pumped fluid. Past studies have resulted in considerable knowledge of jet pump operation, including the governing equations and performance evaluation. Experimental programs have resulted in component efficiency values and have uncovered important factors in hardware design. However, no work has been done to date on the integration of the jet pump into a vehicle-engine system. It was the intent of this analysis to evaluate the utility of jet pumps and to compare them with pressure-fed operation.

It should be pointed out that there are many variations of the basic jet-pump, of which only one is investigated herein. Results of this analysis are not necessarily applicable to other variations.

The gas-drive jet-pump as suggested by previous investigators (Ref. 4, 5, 6) can be represented by the block diagram shown in Fig. 34.

The purpose of each of the components is:

Injector: Mixes incoming high-pressure drive-gas with feedback high-pressure drive-liquid in much the same way a propellant injector mixes fuel and oxidizer.

Drive Nozzle: Accelerates the resulting two-phase mixture to high velocity (typically 600 ft/sec) and low pressure.

Mixer: Introduces the pumped fluid at low pressure, causing a momentum exchange to take place. Velocity is imparted to the pumped fluid.

Separator: The drive gas is allowed to escape while the pumped liquid and feedback liquid are deflected into the diffuser.

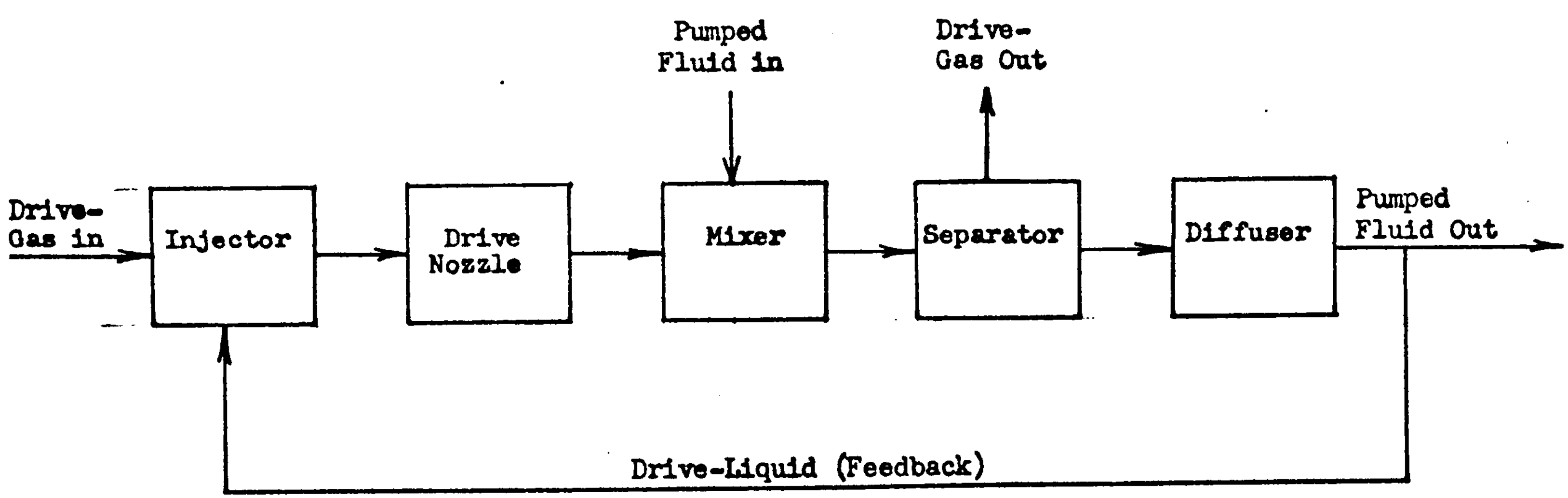


Fig. 34 Gas-Drive Jet-Pump (Schematic)

Diffuser: The dynamic pressure of the fluid stream is converted to static pressure. A portion is then diverted to constitute the feedback liquid, and the remainder flows out through the discharge duct.

The equations for the operation of the jet pump shown in Fig. 34 have been derived in Ref. 6 . With certain assumptions the equation for "gas consumption" is shown to be:

$$\frac{\dot{M}_G}{\dot{M}_D} = \left[\frac{1 + \sqrt{1 - K^2}}{K^2}\right] \left[\frac{P_D - P_S}{\ln P_N/P_S}\right] \left[\frac{W}{R_o\ T_N\ \rho_L}\right] \qquad (1)$$

where

$\dot{M}_G$ = drive-gas flow rate

$\dot{M}_D$ = pumped-fluid flow rate (discharge)

P_D = discharge pressure

P_S = suction pressure (in mixer)

P_N = drive-nozzle pressure (drive-gas inlet-pressure minus injector pressure-drop)

W = molecular weight of drive-gas

T_N = nozzle temperature

ρ_L = pumped fluid density

R_o = universal gas constant (1544 (ft-lb) lb-mole ^oR)

K = momentum recovery factor of jet pump

Furthermore, K can be resolved into:

$$K = K_1 K_2 K_3 K_4 \sqrt{K_D}$$

where

K_1 = nozzle velocity recovery factor

K_2 = mixer momentum recovery factor

K_3 = mass recovery factor

K_4 = separator velocity recovery factor

K_D = diffuser efficiency

The assumptions incorporated in Equation (1) are:

(1) $P_D - P_N \ll 100$

(2) Optimum feedback ratio

both of which are easily attained. The K values account for deviations from ideal processes in the various components of the pump.

A large amount of theoretical and experimental work has been done employing the described jet pump and the above equations. Theoretical efforts have been directed toward rocket engine pumping problems, i.e., specific propellants, (Ref. 6), and toward more sophisticated analytical models of the components (Ref. 7,8). Experimental work has resulted in a knowledge of attainable values for the recovery factors (Ref. 5). While it is not practical to repeat here all of the results of these works, the more important conclusions may be summarized:

(1) A gas-drive jet-pump can be readily started and operated.

(2) Fundamental limitations restrict the momentum recovery factor to K = 0.8 or less. Present technology sets it at K = 0.6.

(3) The separator mass recovery factor K_3 is .90-.95 percent at best, indicating that 5-10 percent of the propellant flow is lost with the drive gas.

(4) Jet-pump gas-consumption rates always exceed those of both pressure and turbopump systems, generally by a factor of 2 to 4.

(5) Jet pumps generally weigh only 1/2 to 1/3 as much as a comparable turbopump.

(6) Jet pumps are inherently simple, and therefore may have a reliability advantage over turbopumps.

These facts indicate that the gas-drive jet-pump will not compare favorably with a pressure-fed or turbopump system, although some advantages do exist. This conclusion must be drawn primarily because of the propellant loss with the drive gas at the separator which has the effect of decreasing specific impulse by 5-10 percent, and because of the corresponding high drive-gas consumption.

To determine how large a payload loss would be associated with the use of a gas-drive jet-pump, a brief mission analysis and comparison was made based on an existing F_2/N_2H_4 system.

A difference in the stage payload capabilities results from the changes in component weights when the pressurized system is converted to use the jet pump. In addition, a decrease in specific impulse results due to the loss of propellant at the separator. These effects are summarized below:

ΔW_{MT} ~ Main tanks

ΔW_{HT} ~ Helium tanks

ΔW_{RG} ~ Residual gases and vapors

ΔW_{JP} ~ Jet pumps

$W_o \; \Delta e^{-\Delta V/g_o I_s}$ ~ Specific impulse

The algebraic sum of these terms is then the payload difference between the two systems. It is seen that changes in payload due to weight savings are independent of mission requirements and engine performance if the initial stage weight, W_o, is held constant. This is a good assumption for small changes in payload and specific impulse, and allows straight-forward evaluation of payload changes. The payload loss due to decreased performance (I_s) is, however, dependent on the specific impulse, velocity increment, and initial weight. For this reason a velocity increment of 20,000 ft/sec was arbitrarily selected. The specific impulse of the jet-pump system was taken to be 6% less than that of the pressure-fed system. Weight changes may be found by calculating the weight of each component based on requirements of the jet pump system and subtracting it from the corresponding weight of the pressure-fed system. A weight savings will then be positive and a loss negative. Briefly, it is found that a weight savings results from the lower main-tank pressure and lesser amount of residual gases and vapors. Weight penalties result from an increased amount of helium and the addition of the pump. The decrease in I_S, of course, results in a payload loss.

By assuming the following reasonable values of the parameters, the weight differences and resulting payload decrease may be found:

$K = 0.5$

$K_3 = 0.94$ (6% propellant loss)

$P_D = 300$ psia

$P_S = 40$ psia

$P_N = 300$ psia

$T_N = 306^{\circ}F$ (oxidizer)

$T_N = 50^{\circ}F$ (Fuel)

$\rho_L = 93.7\ lb/ft^3$ (oxidizer)

$\rho_L = 62.4\ lb/ft^3$ (fuel)

The weight values are calculated to be:

$$\Delta W_{MT} = +318 \text{ lbs}$$

$$\Delta W_{HT} = -530 \text{ lbs}$$

$$\Delta W_{RG} = +25 \text{ lbs}$$

$$\Delta W_{JP} = -13.7 \text{ lbs}$$

$$W_o \ \Delta e^{-\Delta V/g_o I_s} = -233 \text{ lbs}$$

$$PL = -433.7 \text{ lbs}$$

It is readily seen that large penalties are paid for the high drive-gas flow-rates and the propellant loss at the separator. This payload loss is approximately 48%. This loss is, of course, dependent upon the mission ΔV; the percent loss decreases as ΔV decreases, and is approximately 12% when ΔV = 10,000 ft/sec. For the jet pump to show better results, it would be necessary to reduce the drive-gas flow-rate, thereby decreasing weight penalties associated with the helium tank; or to regain the specific impulse lost in propellant escape. The former requires further improvement of jet pump operation, while the latter requires consideration of methods of using the escaping helium and propellant. The suggested techniques for using the separator discharge are:

(1) Dumping it back into the propellant tanks, necessitating an inducer pump in the feed line.

(2) Secondary injection in the thrust chamber.

Improvement of jet-pump operation may result either from improvements of the individual components of the gas-drive jet-pump, or from variations of the basic cycle.

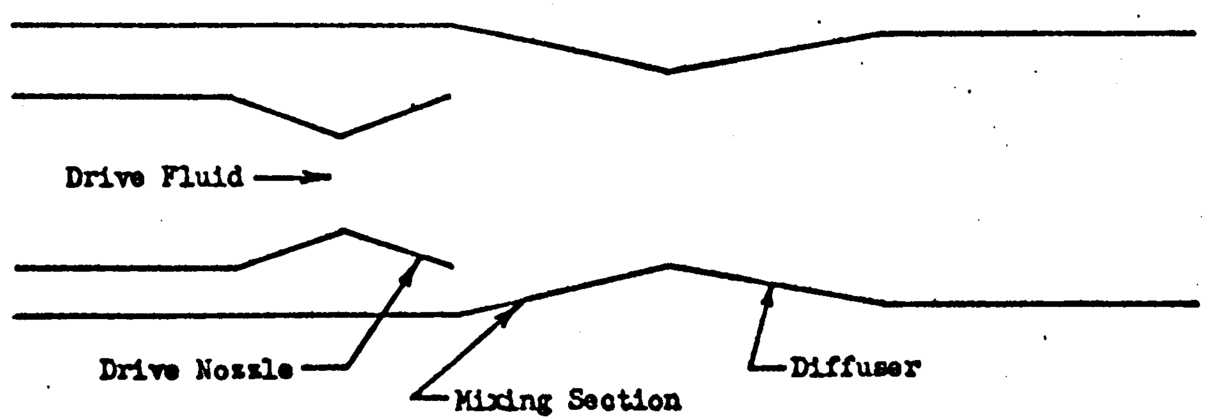
Based on the above analysis, use of the gas-drive jet-pump system instead of a pressure-fed system does not appear attractive. Furthermore, indications are that the payload loss associated with using a jet-pump system instead of a pump-fed (centrifugal or axial) system would be near 20% (ΔV = 10,000).

Jet-Pump-Fed/Pump-Fed System

Analysis of the gas-drive jet-pump (non-condensing) in the preceding section indicated that the major disadvantage of the jet-pump system resulted from the loss of propellant when separating the drive gas from the liquid being pumped. In this section the condensing jet-pump is considered. This is a concept wherein the drive fluid is completely condensed and passes through the pump along with the driven liquid. This eliminates the need for a gas-liquid separator and the loss of liquid associated with it. Figure 35 contains a schematic of this type of jet pump. The primary interest in the condensing jet-pump during this investigation has been for pumping fluorine in a combined jet-pump/conventional-pump LF_2/LH_2 system, wherein the LH_2 would be pumped with a conventional

Figure 35

Condensing Jet-Pump Schematic



turbopump. In the paragraphs that follow, the operating principles of the condensing jet-pump will be reviewed briefly, and the results of a study using such a jet-pump in a combined jet-pump/pump LF_2/LH_2 system presented.

Referring to Fig. 35 , pump operation can be described as follows:

(1) The drive fluid is accelerated as it expands through the drive nozzle.

(2) The drive fluid impinges on the suction fluid, a momentum exchange ensues, and the drive fluid is condensed.

(3) The high-velocity mixture of condensed drive-fluid and suction fluid passes through the diffuser which converts the velocity to static pressure.

The usual method of computing jet pump performance deals with five "processes":

(1) Flow of the suction fluid into the mixing section.

(2) Expansion of the drive fluid.

(3) Momentum exchange between the drive-fluid and the suction-fluid.

(4) Condensation of the drive-fluid by the suction-fluid.

(5) Performance of the diffuser.

The equation governing the suction fluid flow is the standard:

$$v_{SF} = K_1 \sqrt{\Delta P}$$

where

v_{SF} = suction-fluid velocity into the mixing section

K_1 = constant of proportionality

ΔP = pressure drop

Similary, the equation describing the expansion of the drive fluid is well known and has a good theoretical basis; it is:

$$v_{DG} = K_2 \ \eta_N \sqrt{\Delta h}$$

where

v_{DG} = velocity of the drive fluid into the mixing section

K_2 = constant of proportionality

η_N = drive-fluid nozzle efficiency

Δh = drive-fluid enthalpy change through the nozzle

Mathematical description of the momentum exchange represents one of the greater uncertainties associated with jet-pump calculations. The conservation of momentum requires that the following equation be valid in the mixing chamber:

$$\dot{W}_{DF} \, v_{DF} + \dot{W}_{SF} \, v_{SF} = (\dot{W}_{DF} + \dot{W}_{SF}) v_m$$

where

$\dot{W}$ = weight flowrate

v_m = velocity of the mixture of suction fluid and condensed drive-gas as it leaves the mixing section

This equation can be solved for v_m, giving ---

$$v_m = C_i \frac{v_{DF} + \mu v_{SF}}{1 + \mu}$$

where

$$\mu = \frac{\dot{W}_{SF}}{\dot{W}_{DF}}$$

C_i = impact coefficient

The impact coefficient is the big unknown; it is, in effect, a correction factor to bring the theoretical and experimental v_m's into agreement. The condensation-process equation defines a minimum value of mixture ratio, μ, that is required for condensation of the drive fluid; it is ---

$$\dot{W}_{DF} \Delta h_{COND} = \dot{W}_{SF} C_p \Delta T$$

$$\text{or} = \frac{\Delta h_{COND}}{C_p \Delta T} \quad \frac{\dot{W}_{SF}}{\dot{W}_{DF}}$$

where

Δh_{COND} = enthalpy change as the drive fluid condenses.

ΔT = temperature change of the suction fluid from storage conditions to mixing-section conditions.

The process in the diffuser is described by ---

$$P_D = P_{MS} + (1/2 \rho v_m^2) \eta_D$$

where

P_D = static pressure at the diffuser exit

P_{MS} = mixing-section pressure

ρ = density of the mixture

η_D = diffuser efficiency

Previous investigators (Ref. 9 and 10) have analyzed and experimented with condensing jet-pumps, but their efforts have concentrated on pumping efficiencies and discharge pressures obtainable when using vaporized liquid to pump a liquid of the same kind. The effort of this program is different in two respects:

(1) Gas-generator gases are used as the drive fluid,

(2) The jet-pump system is considered from an over-all vehicle standpoint, rather than from the component view.

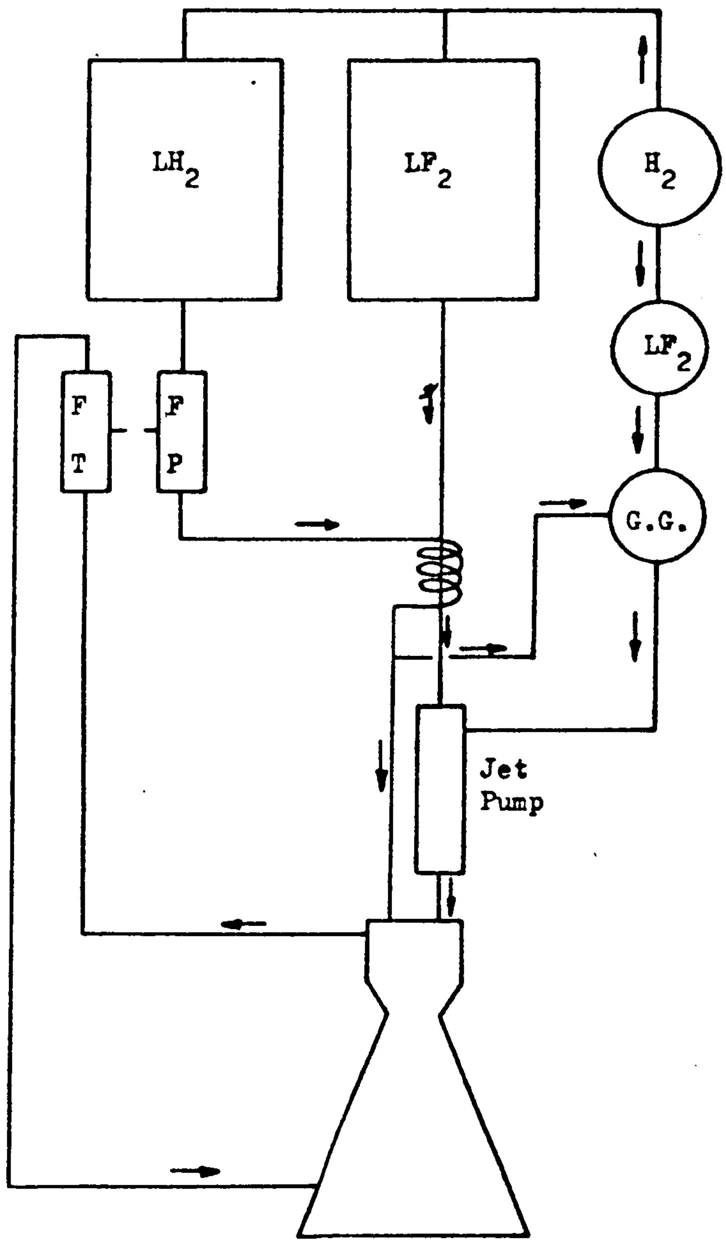
The former consideration eliminates the problem of insufficient energy being available for heating the drive fluid (when vapor is used). The latter consideration focuses attention upon the relative over-all performance of jet-pump systems.

It is apparent from the preceding discussion of the principle of the condensing jet pump that the suction fluid (liquid fluorine) must be sub-cooled before it mixes with the drive fluid (fluorine-rich gas-generator gases), since it must absorb enough heat to condense the drive fluid without becoming vaporized itself.

A major problem associated with this concept is devising a method for sub-cooling the liquid fluorine which will be relatively simple and lightweight. A simple scheme consists of using the liquid hydrogen (which is pumped by a conventional turbopump) to cool the fluorine in a heat exchanger. This is shown schematically in Fig. 36 . A brief analysis was made to estimate the required heat-exchanger surface area. The results indicated the required pressure drop for this concept was excessive. Because heat-exchanger size and weight are strong functions of the particular heat-exchanger design, it was felt that a better approach to use in evaluating the jet-pump system was to consider the weight of the sub-cooling system as a parameter. Thus, the effect on payload could be shown as a function of the sub-cooling-system weight, with the results being applicable to all types of sub-cooling systems.

Fig. 36

Combined Jet-Pump/Pump System



Notes:

1. H_2 Turbopump driven by combustion chamber tapoff gases.

2. F_2 is stored below saturation temperature in main tank and then cooled further in a heat exchanger.

3. G.G. operates at a very high mixture ratio. ($\sim$300)

These results are shown in Figure 37. These curves indicate what the payload advantage is for the combined jet-pump/pump LF_2/LH_2 system over an all pressure-fed system as a function of velocity increment and refrigeration-system (sub-cooling system) weight. The refrigeration-system weight is presented as a fraction of the total jet-pump/pump system inert-weight, where inert weight includes: propellant tanks, structure, pressurant, pressurant storage-bottle, thrust chamber, injector, pumps, lines, valves, and control systems.

The system parameters used for the combined jet-pump/pump system are not necessarily optimum; optimization may result in lighter systems and higher payloads.

The table below describes the two systems compared in Fig. 37.

Jet-Pumped/Pump System:

Thrust	40,000 lbs
Chamber Pressure	90 psia
Area Ratio	130
Propellant Mixture Ratio (F_2/H_2)	10
Gas-Generator Pressure	500 psia
Gas-Generator Mixture Ratio (F_2/H_2)	300
Suction Fluid Velocity	60 ft/sec
Mixing Section Pressure	14.7 psia
Impact Coefficient	.65
Thrust/Vehicle Gross Weight	0.5

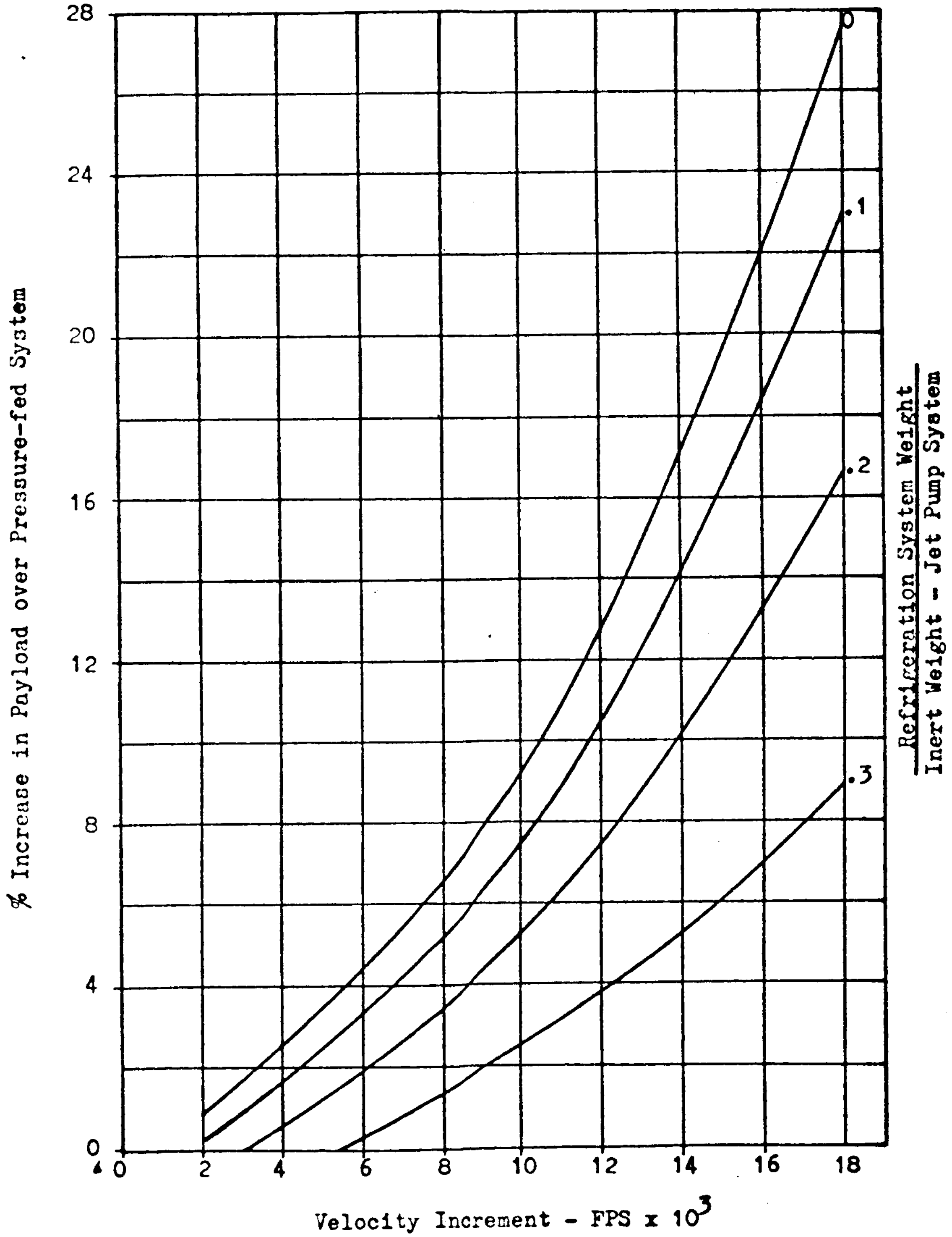


Fig. 37. Payload Comparison - - Jet Pump/Pump-Fed System

Pressure-Fed System:

Thrust	40,000 lbs
Chamber Pressure	90 psia
Area Ratio	130
Propellant Mixture Ratio (F_2/H_2)	10
Thrust/Vehicle Gross Weight	0.5

Subsequent paragraphs discuss the significance of some of the parameters listed above, and what effects their variations have on jet-pump discharge-pressure.

Drive gases for the fluorine jet pump are supplied by the combustion products of a fluorine/hydrogen gas-generator which operates at a high mixture ratio (F_2/H_2). The high gas-generator mixture ratio is desirable for two reasons:

(1) The gas-generator exhaust products contain hydrogen fluoride which is a solid at liquid-fluorine temperatures; the concentration of hydrogen fluoride decreases with increasing mixture ratio, and it is desirable to minimize the concentration of solids.

(2) The temperature and heat capacity (specific heat) of the gas-generator exhaust products both decrease with increasing mixture ratio, thus higher mixture ratios facilitate condensation of the drive gas.

To condense the drive gases, the suction fluid must enter the mixing section in a subcooled condition. The amount of subcooling is dependent upon the heat capacity of the suction fluid, the jet-pump mixture-ratio ($\dot{W}$ suction fluid/$\dot{W}$ drive fluid), and the energy level of the drive gases. A fluorine storage temperature of -355 degrees F was used in this study. Fluorine freezes at -365 degrees F.

Drive-nozzle performance was calculated using Rocketdyne's propellant-performance computer-program. The theoretical-minimum jet-pump mixture-ratio (obtained by an enthalpy balance across the mixing section) was increased by 100% to insure the complete condensation of the drive gases and to allow for the conversion of kinetic energy into heat as the high-velocity drive-gases are slowed in the mixing section. This corresponds to the 33% increase used in Ref. 10. The properties of fluorine and fluorine/hydrogen combustion products dictate the higher number. The diffusion process was assumed to have an efficiency of 90%.

Figure 38 contains a plot of jet-pump discharge-pressure versus mixing-section pressure for parametric gas-generator mixture-ratio. As can be seen the pump-discharge pressure increases with an increase in gas-generator mixture-ratio. A practical upper limit of gas-generator mixture-ratio must be selected from mixture-ratio control and gas-generator injection considerations. The curves in Fig. 38 are for constant gas-generator pressure and constant suction-fluid velocity. It

Fluorine Jet-Pump Performance

V Suction = 60 ft/sec
P Gas Generator = 400 psia
Impact Coefficient = .65

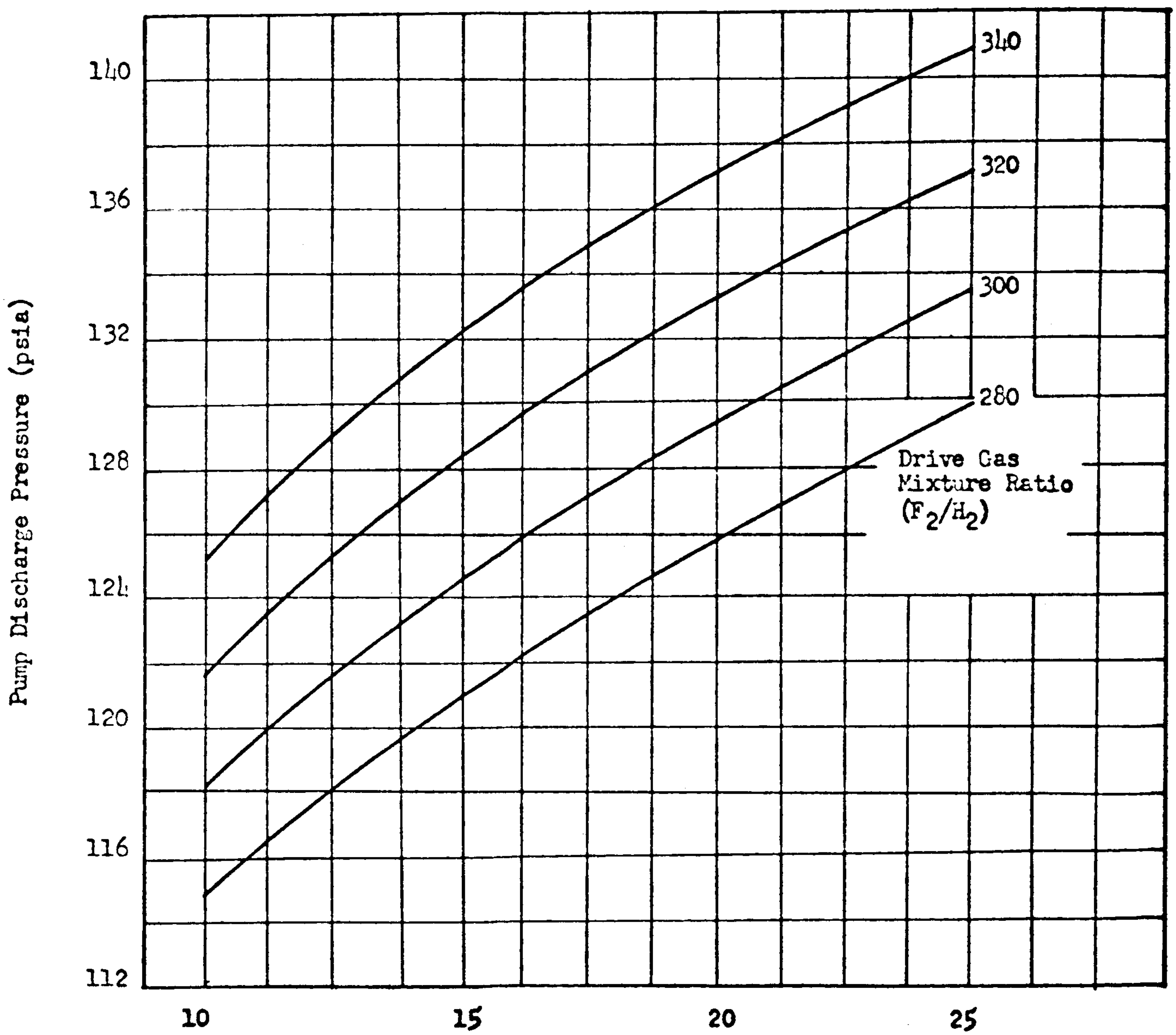


Fig. 38. Jet Pump, Discharge Pressure vs Mixing Pressure and Drive-Gas Mixture Ratio.

should be noted that, for a constant suction-fluid velocity, an increase in mixing-section pressure is accompanied by an even greater increase in tank pressure and, therefore, tank weight.

Figure 39 contains a plot of jet-pump discharge-pressure versus mixing-section pressure for parametric gas-generator pressure. For a given mixing-section pressure, a higher gas-generator pressure requires a greater expansion of the drive gases. The greater expansion results in higher drive-nozzle exit-velocity and, therefore, higher pump discharge pressure. An increase in gas-generator pressure is accompanied by an increase in gas-generator propellant supply-tank pressure. This causes an increase in the weight of both the gas generator and its supply tanks. The gas-generator supply-tanks are quite small, however, and it is possible that gas-generator pressures greater than those presented in Fig. 39 are desirable. It may also be desirable to pressurize the gas-generator supply-tanks with a separate pressurant supply. The main-tank pressurant bottle could then be exhausted to a lower pressure with a resultant saving of pressurant.

Figure 40 contains a plot of jet-pump discharge-pressure versus mixing-section pressure for parametric suction-fluid velocity. A high suction-velocity is desirable because it increases the efficiency of the momentum exchange between suction and drive fluids. This advantage must, however, be weighed against the higher main-tank pressure necessary to obtain the higher suction-fluid velocity.

Fluorine Jet-Pump Performance

V Suction = 60 ft/sec
Impact Coefficient = .65
Drive Gas Mixture Ratio (F_2/H_2) = 300

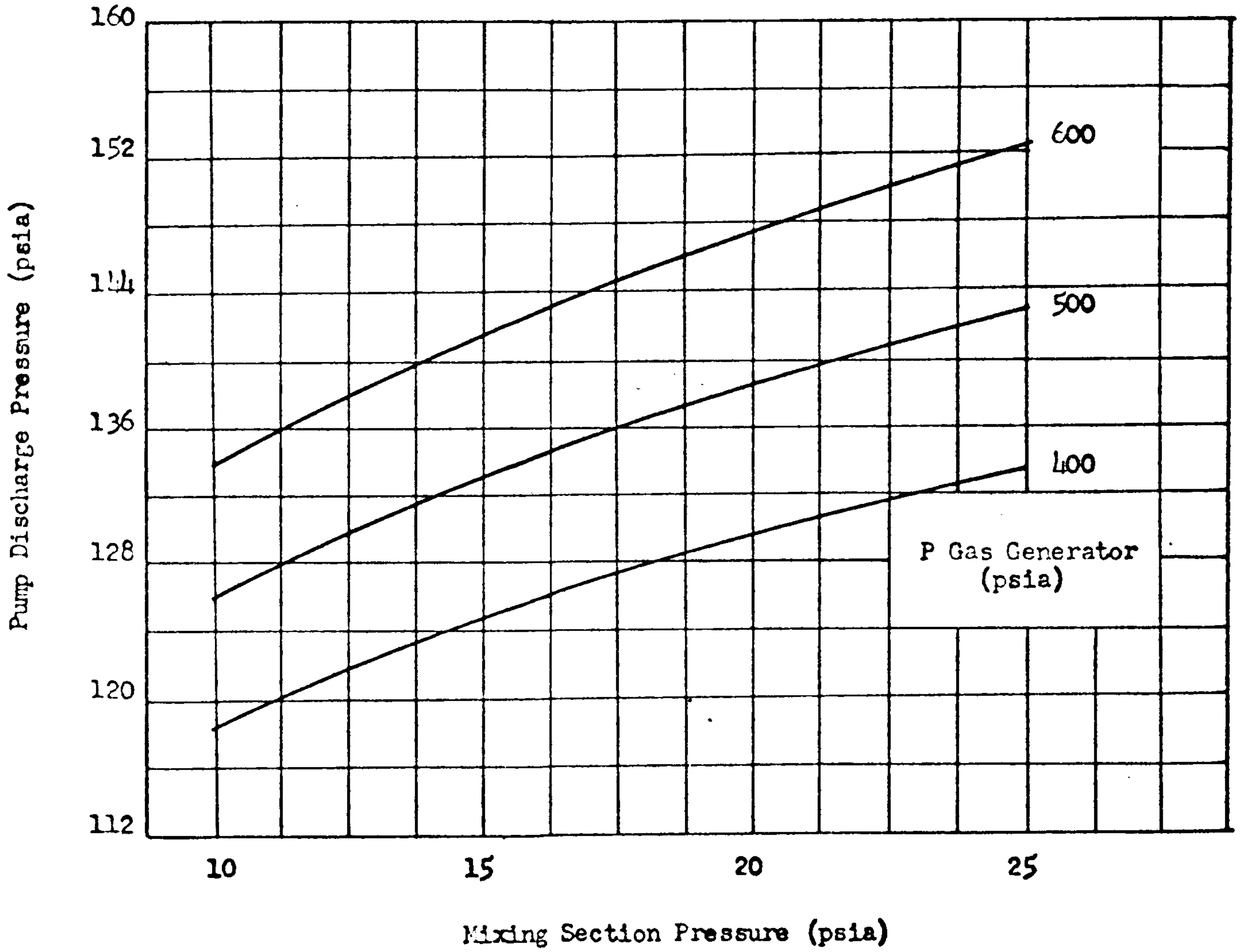


Fig. 39. Jet Pump, Discharge Pressure vs Mixing Pressure and Drive-Gas Pressure.

Fluorine Jet-Pump Performance

P Gas Generator = 400 psia
Impact Coefficient = .65
Drive Gas Mixture Ratio (F_2/H_2) = 300

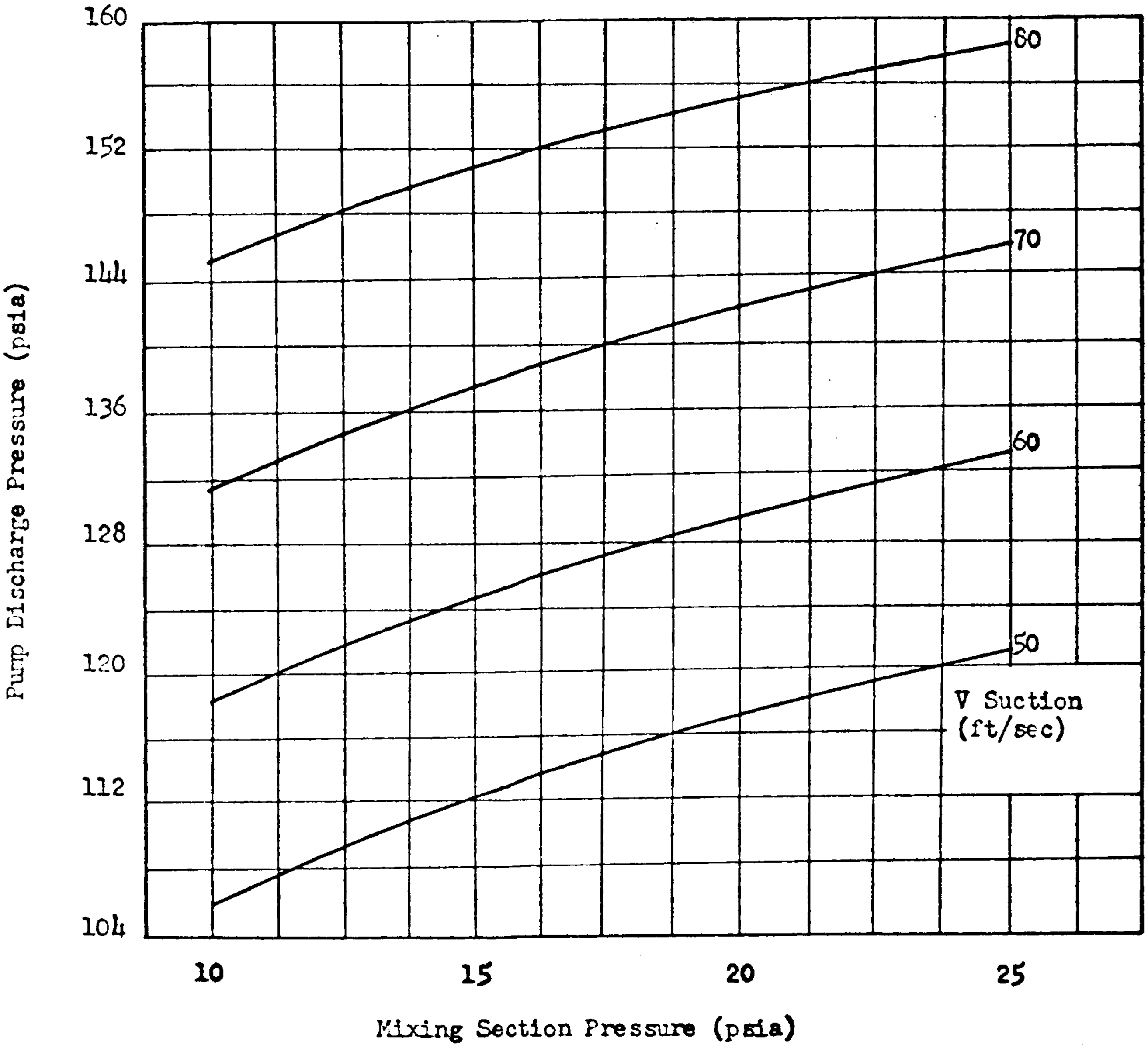


Fig. 40. Jet Pump, Discharge Pressure vs Mixing Pressure and Suction-Fluid Velocity

The momentum-exchange process between suction and drive fluids is of primary importance in jet-pump operation. The final velocity of the mixture of suction and drive fluids is obtained by applying an efficiency or impact coefficient to the momentum-exchange process. The effect of this impact coefficient on pump-discharge pressure is shown in Fig. 41 Previous experiments (Ref. 10) have obtained impact coefficients as high as .75.

Shuttle-Feed Systems

The shuttle-feed concept provides a means of obtaining high chamber pressures without using turbopumps. The shuttle-feed systems considered are high chamber-pressure pressure-fed systems using O_2/H_2; the propellants are expelled from small high-pressure tanks by pressurizing with catalytically ignited O_2/H_2. This section describes their operation and presents an analytical evaluation of the concepts. The three systems considered are:

(1) Non-venting system (Fig. 42)

(2) Non-venting system using pump-tank heat exchangers (Fig. 43)

(3) Vent-to-main-tank and ambient system (Fig. 44)

Fluorine Jet-Pump Performance

V Suction = 60 ft/sec
P Gas Generator = 400 psia
Drive Gas Mixture Ratio (F_2/H_2) = 300

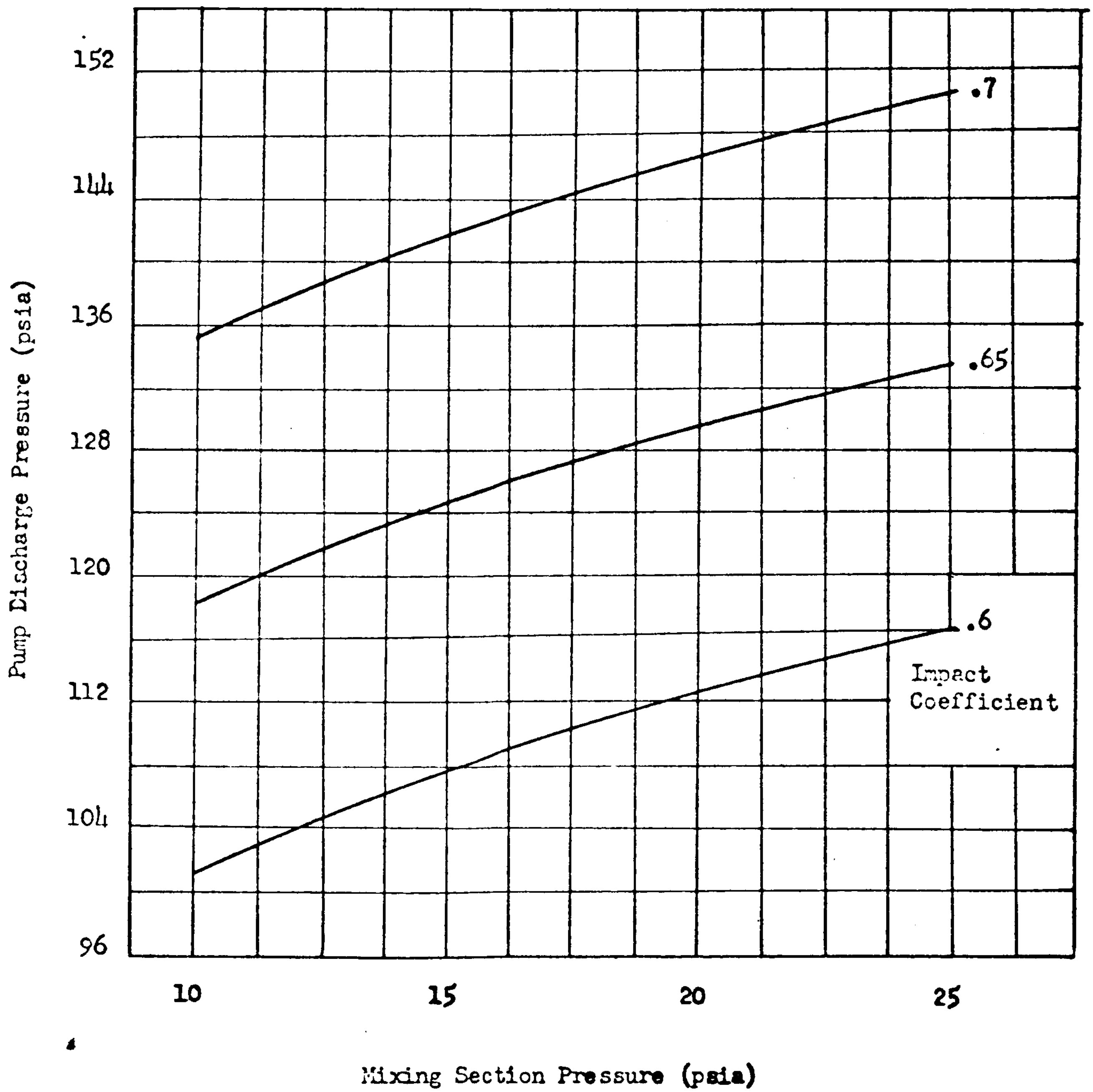


Fig. 41. Jet Pump, Discharge Pressure vs Mixing Pressure and Impact Coefficient

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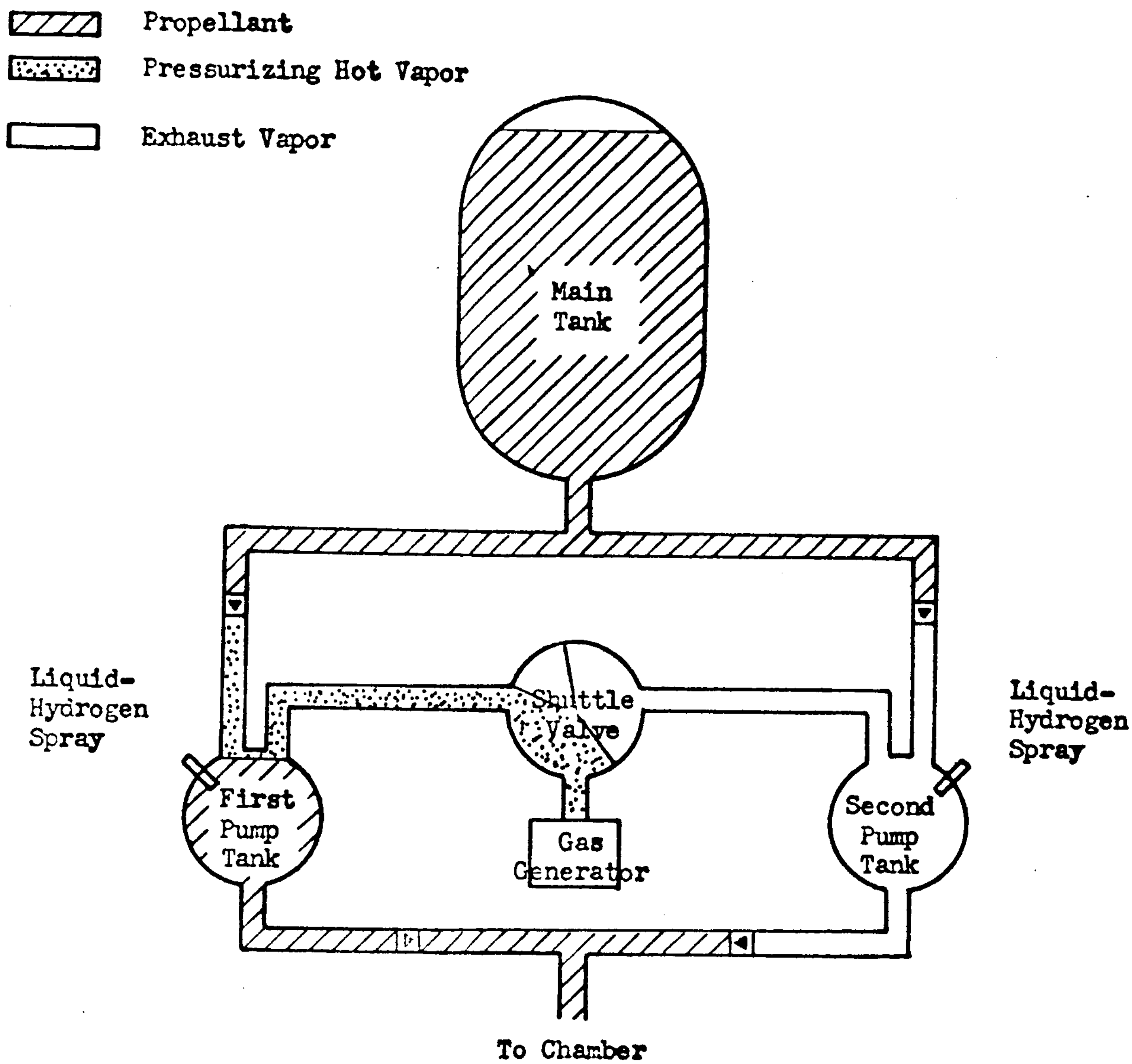


Fig. 43

"Topping" Shuttle-Feed Schematic

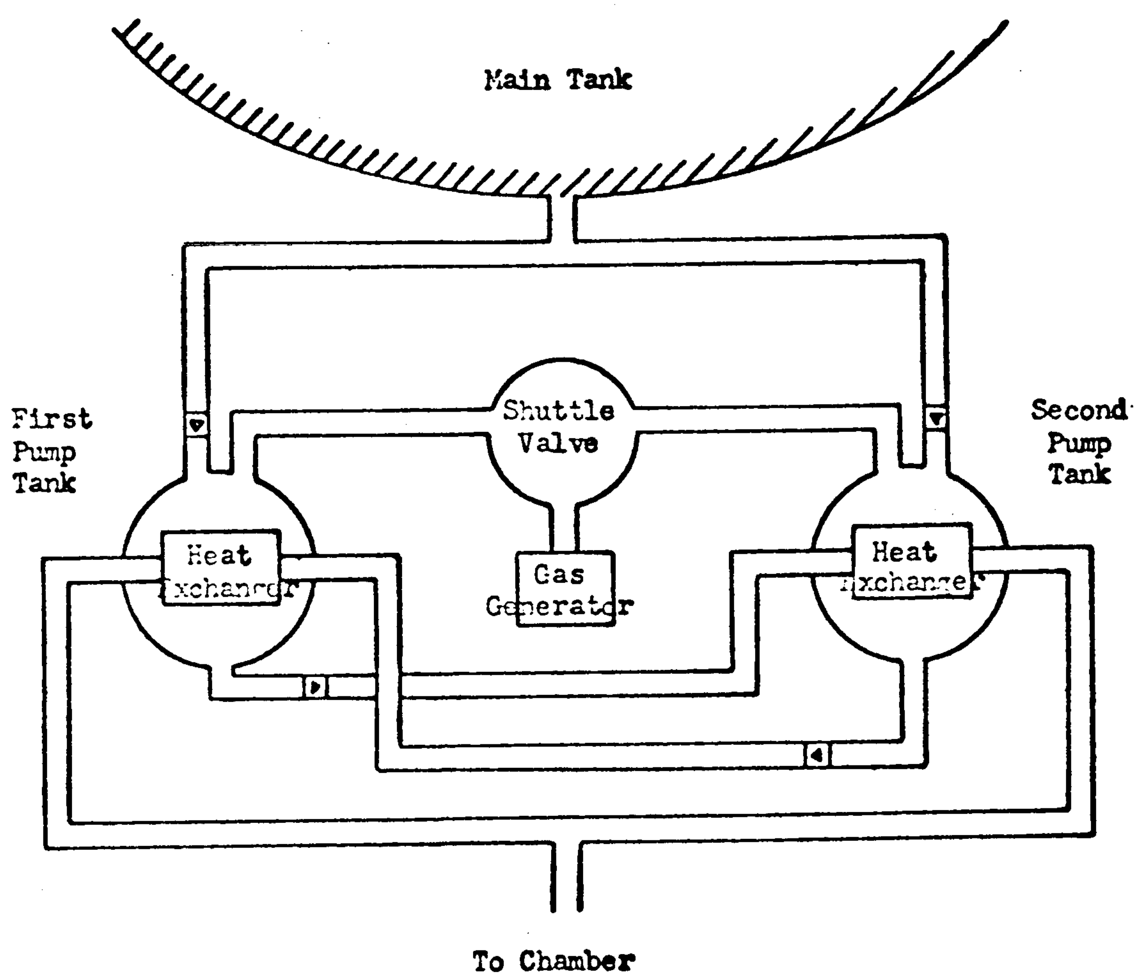
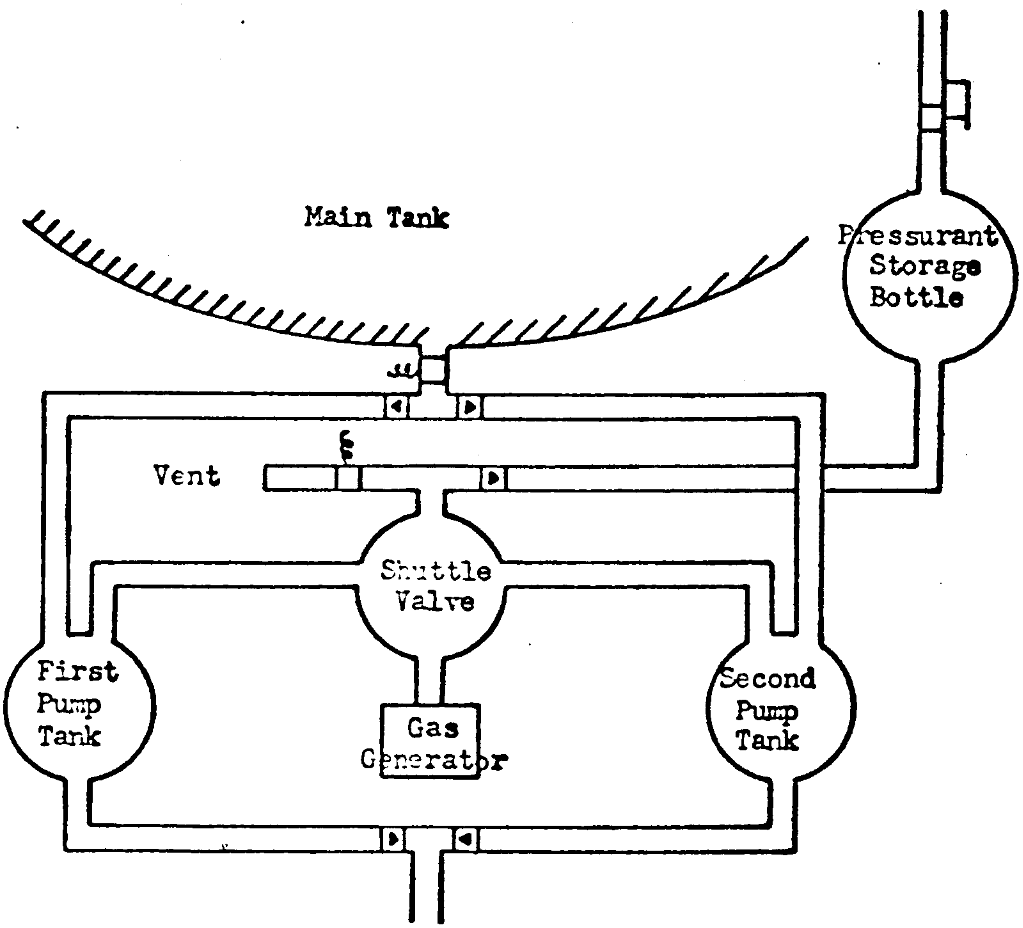


Fig. 44

Shuttle-Feed Schematic



All three of these systems have the following features in common:

(1) A gas-generator (catalytically-ignited O_2/H_2 at 960°R)

(2) Shuttle valve --- to control gas-generator flow

(3) Check valves in the lines from the main tank --- to stop reverse flow when pump tanks are being pressurized by the gas-generator.

In the following discussion the first two systems are shown to be infeasible; and the third is shown to be feasible. Operating characteristics of the feasible configuration (system 3) are discussed in more detail, and it is shown that approximately 4% of the pumped hydrogen must be vented overboard during system operation.

Operation of these three systems is similar and can be described as follows (starting the cycle with both pump tanks filled):

(1) Pressurant from the gas-generator expels propellant from the first tank.

(2) Near depletion of the first tank, the shuttle valve is actuated to direct pressurant to the second tank.

(3) The pressure trapped in the first tank is reduced (each system uses a different method) to allow the pump tank to be refilled from the main tank.

(4) Near depletion of the second tank, the shuttle valve is actuated to direct pressurant to the first tank --- thus starting a new cycle.

The methods employed in each of these systems to reduce the pump-tank pressure prior to refilling will now be discussed.

The non-venting system (system 1) sprays LH_2 into the tanks to cool the pressurant; this LH_2 comes from the same high-pressure source which is used for the gas-generator. Results of the analysis for this system (presented later in this section) indicate that the pressure cannot be reduced sufficiently to allow propellants to flow from the main tank.

The non-venting system using pump-tank heat exchangers (Fig. 43) passes the pumped propellant from one pump tank through a heat exchanger to cool the hot pressurant in the other pump tank. The cooling of the pressurant decreases the pump-tank pressure from 400 psia to below the main tank pressure of 50 psia, thus allowing the pump tank to be refilled with liquid propellant.

The third system (main-tank and ambient vent) reduces the pump-tank pressure by first venting to the main tank, then venting to ambient pressure (vacuum) to further reduce the pressure. Analysis indicates this concept could also work.

Subsequent paragraphs present the analyses and evaluations of these three systems.

The following symbols will be used in the calculations below:

f (subscript) = saturated liquid

v (subscript) = saturated vapor

T = temperature (^{o}R)

P = pressure (psia)

v = specific volume (ft^3/lb)

h = enthalpy (btu/lb)

h_{fg} = heat of vaporization (btu/lb)

c_p = specific heat capacity (btu/lb ^{o}R)

In these calculations the main tank pressure is assumed to be 50 psia. The pressurizing gas in the pump tanks is at a pressure of 600 psia and a temperature of $400^{o}R$ where:

$v = 3.65\ ft^3/lb$

$h = 1298.6$ btu/lb

After the propellant has been expelled from the pump tank in Method (1), an attempt is made to cool the hydrogen vapor to a saturation pressure of 40 psia where:

T (sat) = $43.5^{o}R$

h_g = 88.0 btu/lb

h_{fg} = 178.6 btu/lb

v_g = $4.72\ ft^3/lb$

The sprayed hydrogen is assumed to be at a saturation pressure of 15.0 psia where:

$T = 36.6^{\circ}R$

$c_p = 2.25$ btu/lb $^{\circ}R$

$v_f = 0.2266$ ft^3/lb

The weight of the hydrogen vapor that is left in the pump tank after the propellant has been expelled is:

$$m = \frac{\text{Volume of Tank}}{\text{v at 600 psia and } 400^{\circ}R} = \frac{1.0}{3.65} = 0.274 \text{ lb}$$

The weight of the liquid hydrogen that must be sprayed into the pump tank so the hydrogen in the tank will be a saturated vapor at 40 psia is:

$$x = \frac{m\,(h(600 \text{ psia}, 400^{\circ}R) - h_g(40 \text{ psia}))}{h_{fg}(40 \text{ psia}) + (c_p(15 \text{ psia}))\,(T_{sat} \text{ at } 40 \text{ psia} - T_{sat} \text{ at } 15 \text{ psia})}$$

$$= \frac{0.274(1298.6 - 88.0)}{178.6 + (2.25)\,(43.5 - 36.6)} = 1.71 \text{ lb}$$

Therefore the total weight of the saturated hydrogen vapor in the pump tank at 40 psia is:

$x + m = 1.98$ lb

But the volume that 1.98 lb of saturated hydrogen vapor at 40 psia would occupy is:

$$V_{Required} = 4.72(1.98) = 9.35 \text{ ft}^3$$

Since the tank volume is only 1.0 ft^3, it would be impossible to cool the hydrogen vapor in the tank to a saturation pressure of 40 psia.

Similar calculations for a pressuring pressure of 100 psia, i.e., for a lower chamber pressure, instead of 600 psia are as follows:

$$m = \frac{1.0}{21.40} = 0.0468 \text{ lb}$$

$$x = \frac{0.0468(1297.2 - 88.0)}{178.6 + 15.5} = 0.291 \text{ lb}$$

$$x + m = 0.338 \text{ lb}$$

$$V_{Required} = 4.72(0.338) = 1.6 \text{ ft}^3$$

Therefore, even if the pressurizing pressure is reduced to 100 psia, it would be impossible to depressurize the pump tanks to 40 psia

The following calculations using hydrogen as the propellant and an O_2/H_2 gas generator show that the system 2 is infeasible. These calculations are based on a heat-exchanger efficiency of 100%.

In Case (1), it is assumed that no mixing occurs between the cooled hydrogen vapor in the pump tank (Point C in Fig. 45) and the entering liquid hydrogen (Point F). For Case (2) it is assumed that complete mixing occurs between the cooled hydrogen-vapor and the liquid hydrogen.

For Case (1), the hydrogen vapor at Point A will be cooled by the heat exchanger to Point B in an isometric process and then to Point C in an isobaric process (to an assumed temperature of $60^{\circ}R$). The part of the pump-tank volume that could be filled by liquid hydrogen is 1.0 - 0.085/0.170 = 0.50 or 50%.

For Case (2), the incoming liquid hydrogen at Point F will mix with the hydrogen vapor at Point C with the final state point of the mixture being Point G. Therefore:

$$W_F = W_C(h_C - h_G)/h_{FG} = 0.265\, W_C \text{ lbs}$$

If no mixing occurs, the volume that 1.0 lb of W_C and 0.265 lbs of W_F would occupy is:

$$V_1 = \frac{W_C}{C} + \frac{W_F}{F} = 5.95 \text{ ft}^3$$

If mixing occurs:

$$V_2 = \frac{W_C \quad W_F}{G} = 4.81 \text{ ft}^3$$

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CHECKED BY:	Fig. 45	REPORT NO.
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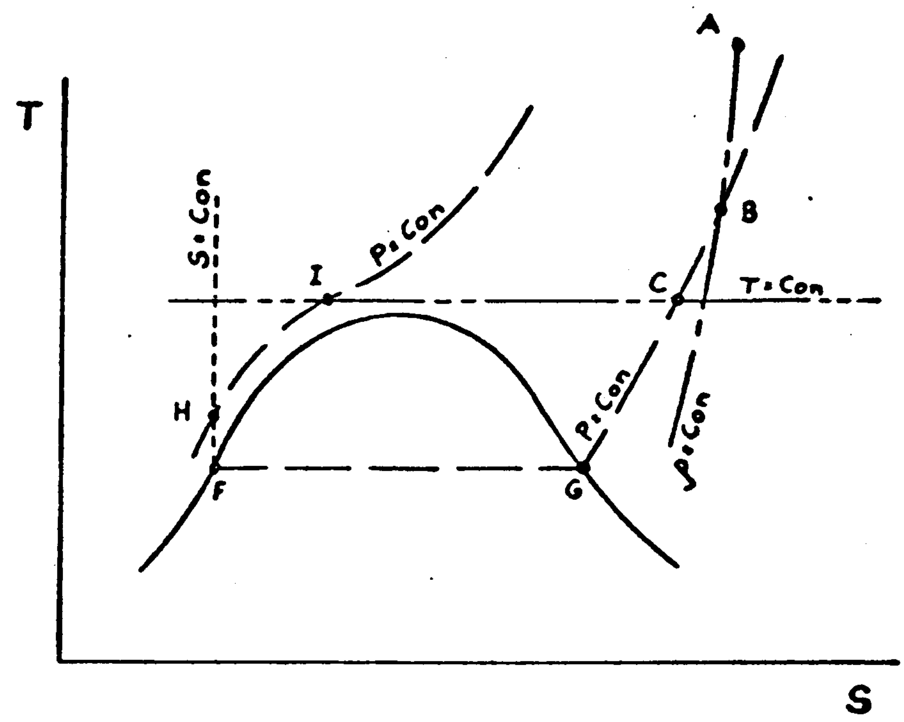


Point A

P = 400 psia
T = 860°R
= 0.085 lb/ft^3
h = 2930 btu/lb

Point B

P = 50 psia
T = 110°R
= 0.085 lb/ft^3

Point C

P = 50 psia
T = 60°R
= 0.17 lb/ft^3
h = 134.1 btu/lb

Point F

P = 50 psia (saturated liquid)
T = 45.4°R
= 4.0 lb/ft^3
h = 84.5 btu/lb

Point G

P = 50 psia (saturated vapor)
T = 45.4°R
= 0.263 lb/ft^3
h = 88.3 btu/lb

Point H

P = 400 psia
T = 48.5°R
= 4.14 lb/ft^3
h = 68.6 btu/lb

Point I

P = 400 psia
T = 60°R
= 3.46 lb/ft^3
h = 26.2 btu/lb

Therefore, if mixing occurs, 1.0 - (0.5)(4.81)/5.95 = 0.596, or 59.6% of the tank could be filled with liquid hydrogen. It now remains to determine if the amount of liquid hydrogen passing through the heat exchanger is sufficient to cool the pressurant to 60 R as was assumed in the above calculations. The liquid hydrogen at 400 psia (Point H) is heated in the heat exchanger to Point I. The hydrogen coolant required for the heat exchanger is:

$$W_H = W_A(h_A - h_C)/(h_H - h_I) = 5.60 \text{ lbs}$$

But only (0.596)(4.0) = 2.38 lbs of liquid hydrogen flows in one tank cycle, therefore the hydrogen vapor at Point A cannot be cooled to Point C.

System 3 (Fig. 14) reduces the pump-tank pressure by first venting to the pressurant storage, and then to the ambient environment. A valve (to prevent propellant from the main tank from flowing overboard during venting) is to be closed when the vent valve is opened; this valve could be mechanically linked to the vent valve.

Two methods have been used to estimate the amount of main-tank pressurant required during operation of system 3. The following additional symbols will be used in this discussion:

x = weight of pressurant required (lbs)

y = weight of vaporized propellant (lbs)

z = weight of liquid propellant

Tank pressure = 50 psia

In Method 1, equilibrium is not obtained in the propellant tank (i.e., $y = 0$). When $T_x = 400°R$

$$V_x = 42.69\ ft^3/lb$$

$$x = 1.0/v_x = 0.0234\ lbs$$

In Method 2, y pounds of propellant vaporize to cool the hot pressurant x to the saturation temperature of z.

Now

$$(x + y)v_y = 1.0 + yv_z$$

$$x(h_x - h_y) = y(h_{fg})$$

Therefore, since $yv_z \approx 0.0$

$$x = \frac{1.0}{v_y(1.0 + (h_x - h_y)/h_{fg})}$$

When $T_x = 400°R$

$$V_y = 3.80\ ft^3/lb$$

$$h_x = 1298.6\ btu/lb$$

$$h_y = 88.3\ btu/lb$$

$$h_{fg} = 172.8\ btu/lb$$

$$x = 0.0328\ lbs$$

$$y = 1.0/v_y - x = 0.230\ lbs$$

Results of both methods are shown in Fig. 46. Assuming that the incoming pressurant temperature is 300°R, the weight of the required pressurant would be between 0.030 lbs to 0.046 pounds depending on the measure of equilibrium reached in the main tank. Figure 46 shows that the incoming pressurant should be cooled to as low a temperature as possible to minimize the amount of hydrogen vented overboard, and vaporized propellant. This could be accomplished by using a heat exchanger on either side of the pressurant storage-tank.

Because the attractiveness of this concept is greatly dependent upon cycle frequency (for higher frequencies mean that smaller and lighter pump-tanks can be used), a brief investigation was made to determine cycle rates.

The cycle rate and pump-tank volumes determine the obtainable propellant flowrate of the system. The cycle rate for a set of two tanks is:

$$CR = 1.0/((t_1 + t_2 + t_3)\ DP)$$

Where:

t_1 = time required to refill pump tank with propellant (sec) = Vol/AV

Vol = volume of one pump tank (ft^3)

A = cross-sectional area of 6"D pipe (0.196 ft^2)

V = average velocity of refilling propellant (20 ft/sec)

t_2 = time required to pressurize pressurant storage-tank = 0.1 Vol (sec)

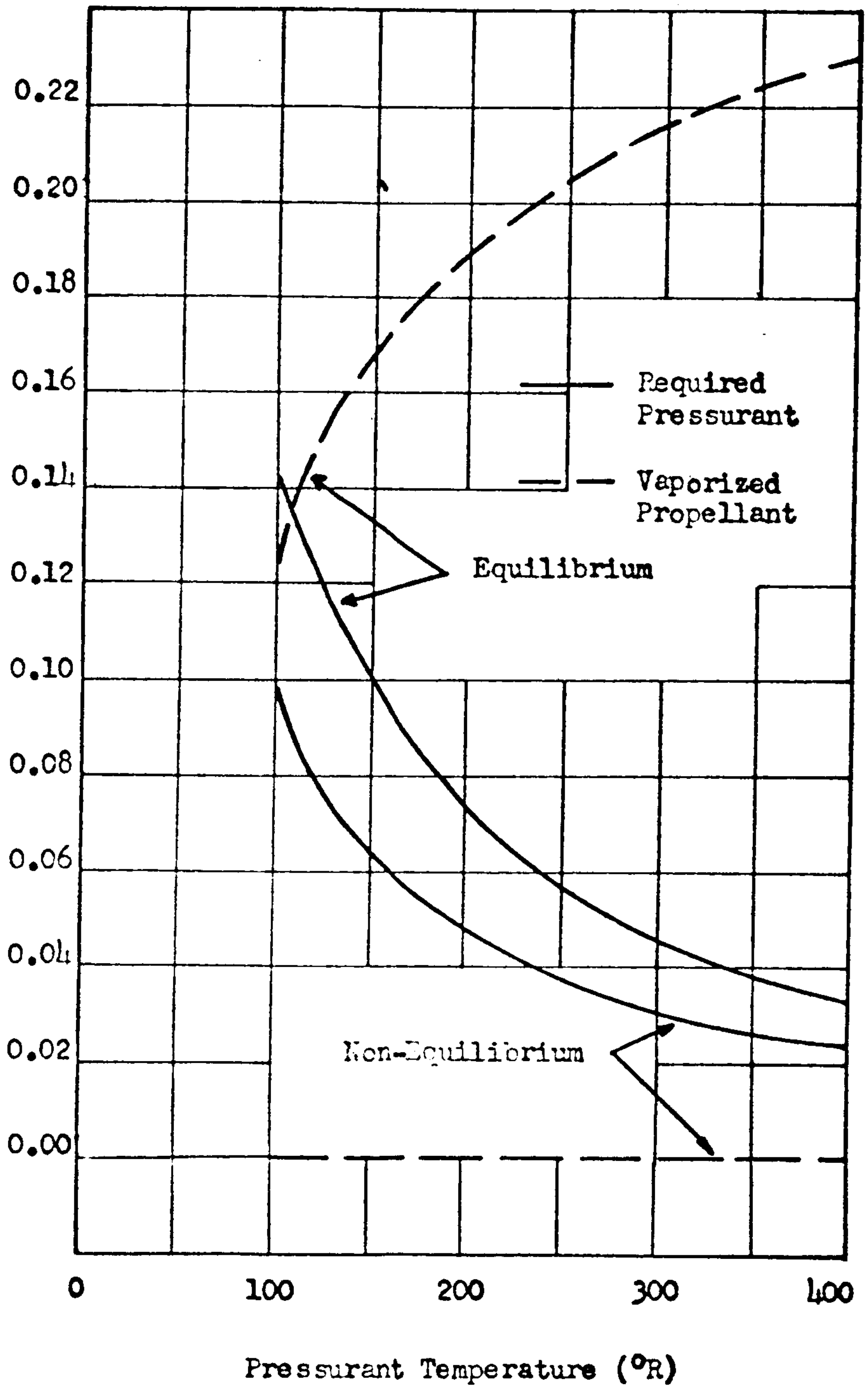


Fig. 46. Shuttle-Feed, Pressurant Weight and Temperature.

t_3 = time required to vent pump-tank to atmosphere = 0.1 Vol (sec)

DP = delay period adjustment = 1.1

Results are shown in Figure 47. These results indicate relatively large inlet lines (and valves) are required if pump-tank size and cycle rate are to be reasonable.

In system 3 it was assumed that no heat transfer occurred between the incoming hot pressurant and the liquid propellant in the pump tank. A pressurant final temperature was assumed and the amount of pressurant required to empty a full pump-tank was the weight of the saturated vapor that would occupy the pump-tank volume.

In the following paragraphs, the results of the variation of the gas-generator combustion temperature and the assumed saturated hydrogen-vapor (pressurant) temperature were calculated considering heat transfer between the pressurant and propellant.

A schematic of the system analyzed is shown in Figure 48. Results in Table 8 were for a 540 deg. R gas-generator combustion temperature and an assumed hydrogen vapor temperature of 400 deg. R. Figure 49 shows that any change in the gas-generator combustion temperature has little effect on the system results, if the assumed saturated hydrogen-vapor pressure (after the pump tank is emptied) is constant. However, the weight of hydrogen vapor that must either be vented to the main propellant tank

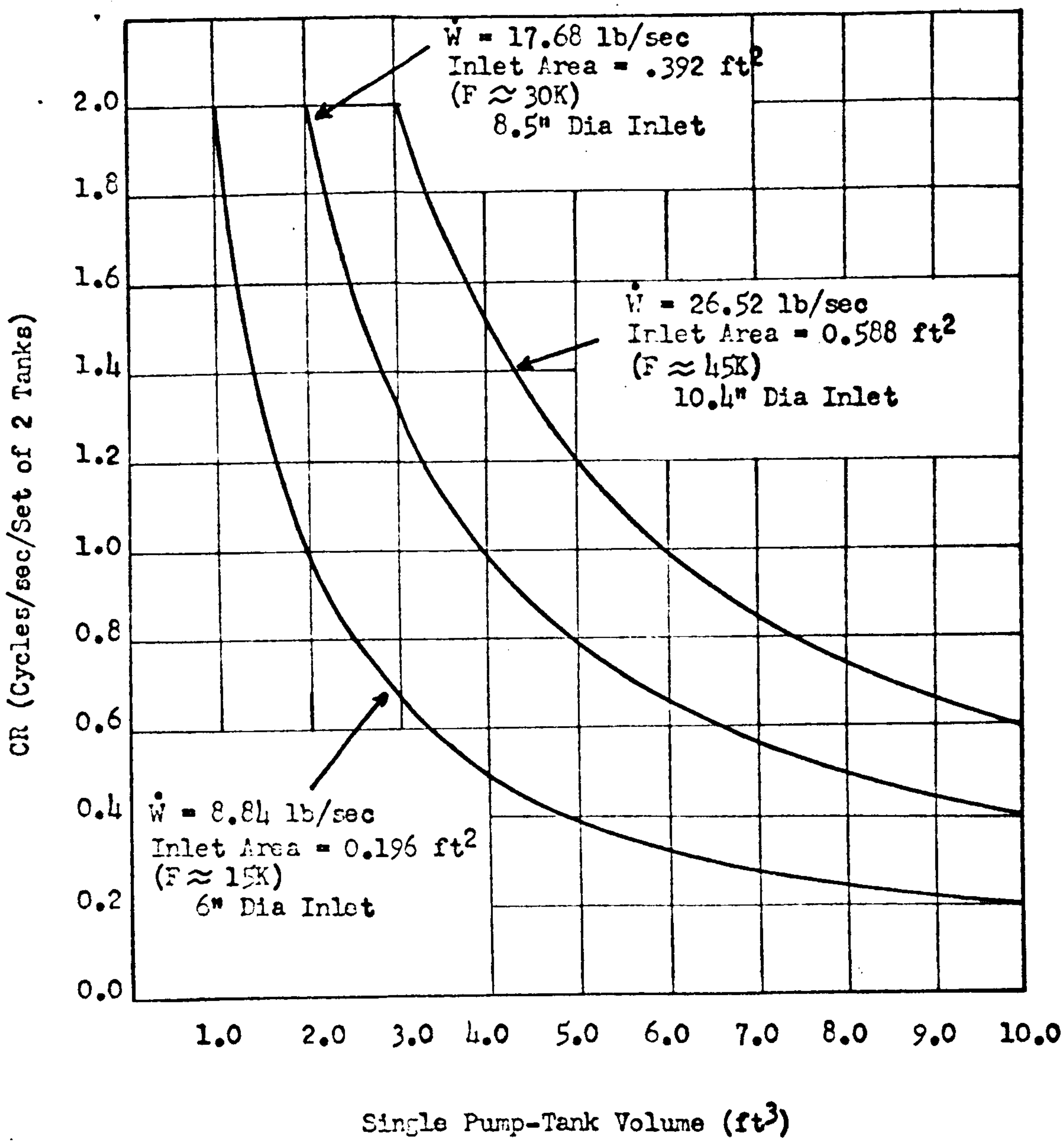


Fig. 47. Shuttle-Feed, Cycle Rate and Pump-Tank Volume.

Figure 48

Schematic for Shuttle Tank System Balance

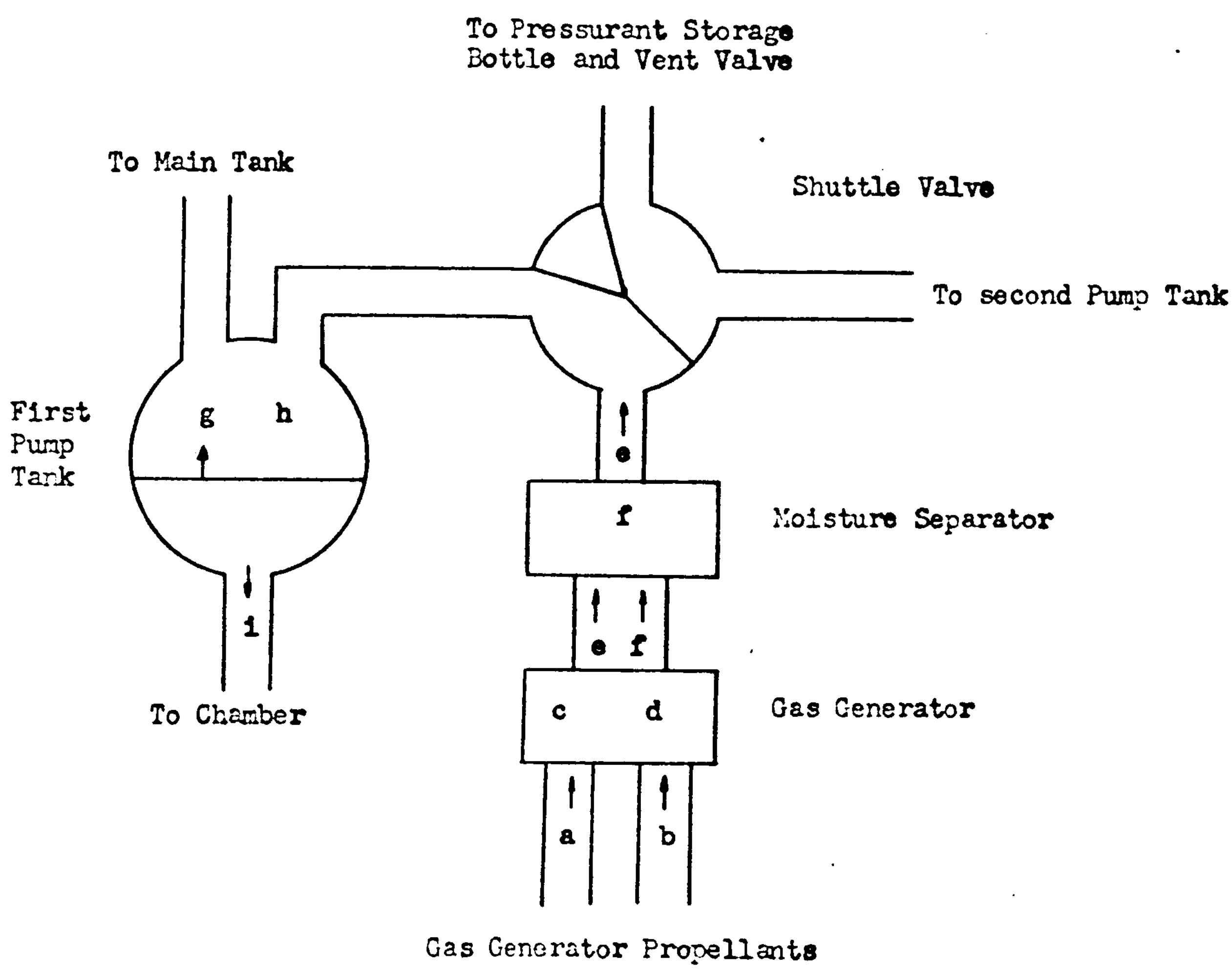


TABLE 8

Identification and Results (for $T_c = 540^oR$, $T_h = 400^oR$)

Symbol	Identification	Weight (lb)
a	Liquid hydrogen entering gas generator	0.2062
b	Liquid oxygen entering gas generator	0.0515
c	Hydrogen burned in gas generator	0.0064
d	Oxygen burned in gas generator	0.0515
e	Hydrogen pressurant required to empty pump tank	0.1998
f	H_2O leaving gas generator	0.0580
g	Liquid hydrogen in pump tank vaporized by hot pressurant	0.0742
h	Saturated hydrogen vapor remaining in pump tank after the propellant has been emptied	0.2740
i	Propellant pumped in one pump tank cycle	4.339
j	Propellant required for gas generator	0.2577

——— T_c = 540°R

— — T_c = 760°R

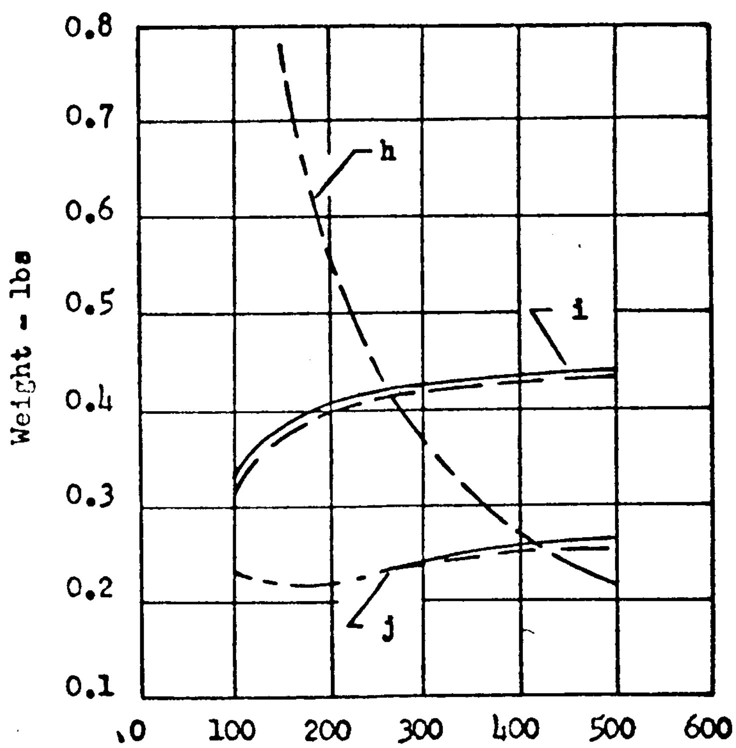
— - — T_c = 540°R and 760°R

T_c = Gas Generator Combustion Temperature

h = Saturated Hydrogen Vapor in Pump Tank

i = Propellant Pumped in One Pump Tank Cycle

j = Propellant Required for Gas Generator



T_h = Saturated Hydrogen Vapor Temperature (°R) in Pump Tank

Figure 49. Shuttle-Feed System, Effect of Pressurant Temperature.

or vented overboard depends greatly on the assumed saturated hydrogen-vapor temperature. It can be concluded that the pressurant temperature in the pump tank should be maintained at as high a value as possible compatable with system design for optimum performance of the system.

The equations used in making these determinations are presented below. A mass balance of the gas generator gives:

$$a = c + e$$

$$MR_{GG} = b/(c + e)$$

$$j = a + b$$

The stoichiometric mixture ratio for complete combustion of hydrogen and oxygen is 8.0.

$$MR_{ST} = 8.0 = d/c = b/c$$

$$MR_{GG} = 8.0\ c/(c + e)$$

$$c = e\ MR_{GG}/(8 - MR_{GG})$$

$$f = c + d$$

An energy and mass balance for the pump tank gives:

$$h = e + g$$

$$e\ (h_e - h_h) = g\ (h_h - h_i)$$

As an example, each pump tank was assumed to be 1.0 ft^3.

$$(e + g)\, v_g = 1.0$$

$$g = 1.0/(v_h\,(1.0 + (h_h - h_i)/(h_e - h_h)))$$

$$e = 1.0/v_h - g$$

$$i = 1.0/v_i - g$$

DESIGN INVESTIGATIONS

The design investigations have been directed primarily at implementing physical integration of engine-system components. In some measure, this has required the consideration of over-all engine-system configurations. Methods of integrating various components and subsystems of the basic systems defined earlier have been investigated. Preliminary-design layouts and/or conceptual sketches of the evolved concepts are presented in this section, and their applicability within the scope of this program is indicated.

Systems

Preliminary-design layouts have been made for a number of the selected basic systems, both spacecraft and boosters. The purpose of these layouts was to provide direction for component integration, and in some cases, to illustrate a particular system concept considered in the analytical investigations.

Spacecraft. Preliminary-design layouts have been made for representative system configurations using conventional and advanced nozzles. These include single-shaft and dual-shaft turbopump systems.

Conventional Nozzles. Figures 49 and 50 contain layouts for spacecraft systems using conventional bell nozzles.

The system shown in Figure 49 is for a 40K NTO/50-50 engine, and is essentially based on system 302 (see Fig. 14 for schematic). This figure depicts the over-all configuration associated with the following integration concepts:

(1) The 3-leg gimbal system, which integrates gimbal, thrust structure and propellant inlet ducts.

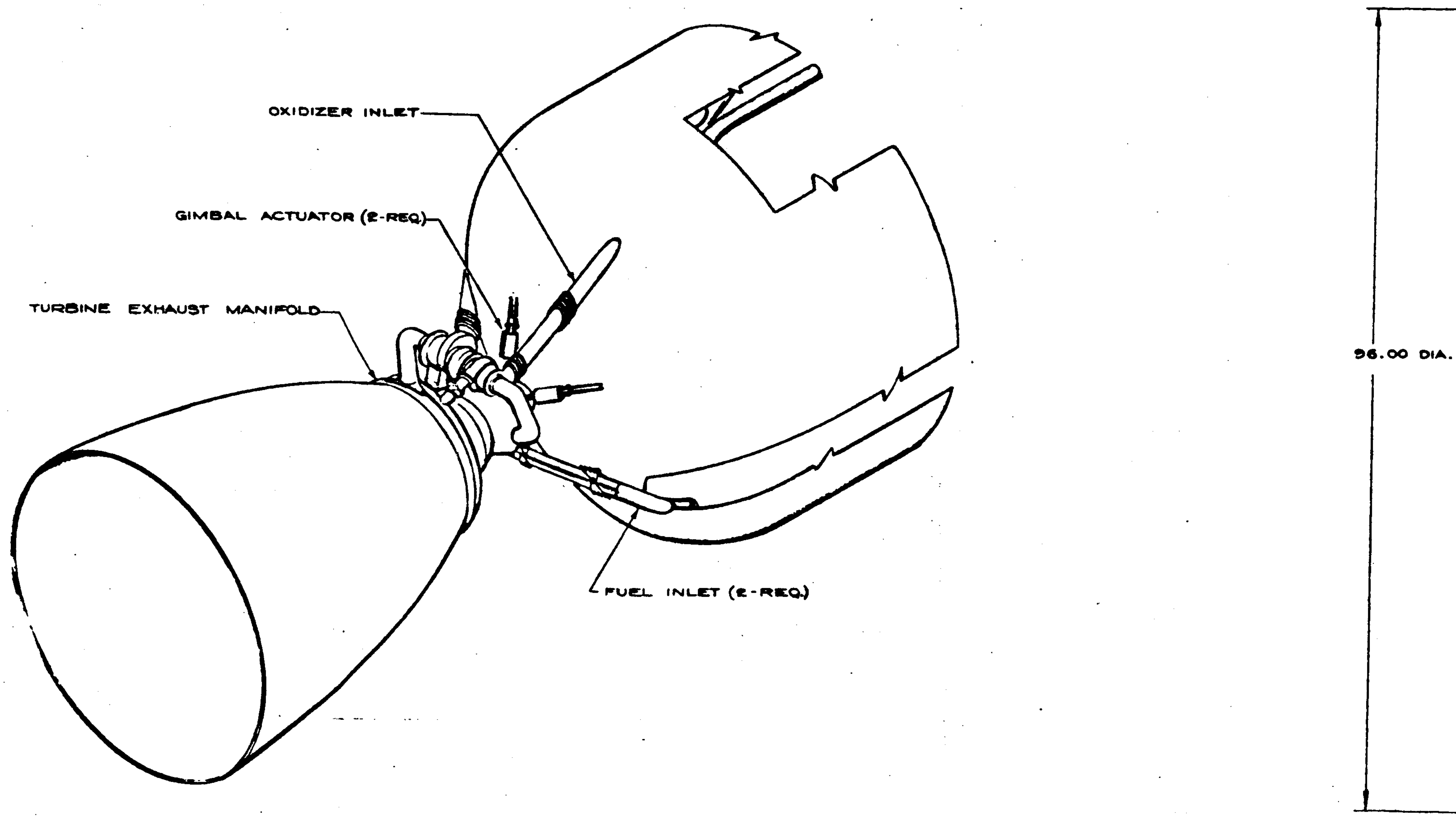
(2) Thrust-chamber tapoff for turbine gases.

(3) Integrated gas-generator start-system which integrates, start tanks, gas-generators and several valves (Fig. 73 or 74).

(4) Integral main-propellant valves and oxidizer igniter-valve (Figs. 61 and 62).

The thrust from the engine is transmitted to the vehicle through the three-leg gimbal system. The three-leg gimbal system consists of the thrust chamber and struts with pinned or ball joints at each end. As shown in Fig. 49, the system allows the engine to be gimbaled about a point which is approximated by the initial intersection of the struts.

[redacted]



PERSPECTIVE

FOLD-OUT #1

[redacted]

DESIGN PARAMETERS

ENGINE THRUST ——— 40 K
PROPELLANTS ——— NTO / 50-50
THRUST CHAMBER ∈ ——— 184
CHAMBER PRESS. ——— 1000 PSIA.

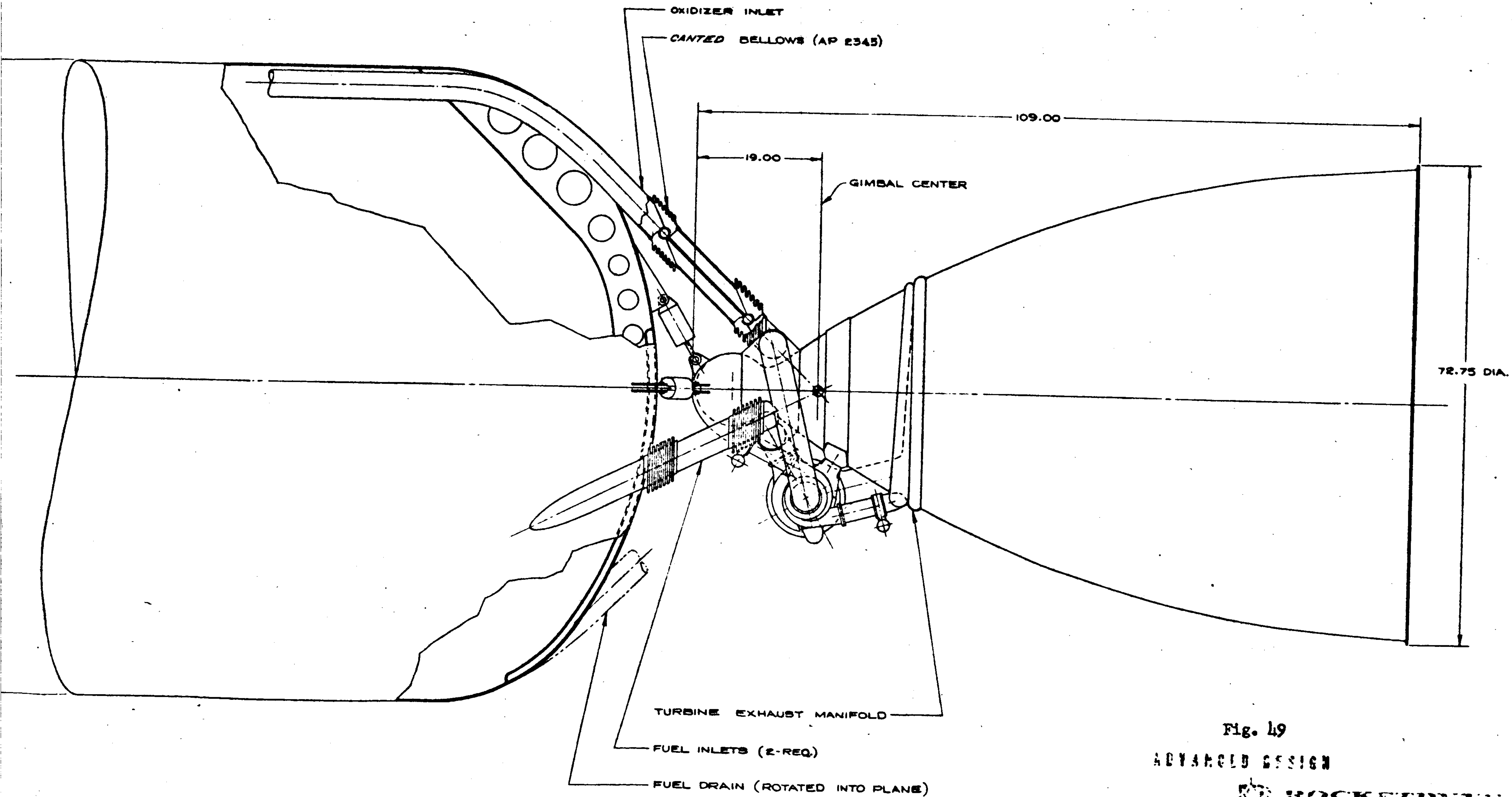


Fig. 49

ADVANCED DESIGN

ROCKETDYNE

40 K ROCKET ENGINE
WITH 3-LEG GIMBAL SYSTEM

J. C. Cowell

1/8 K SIZE JAN. 21, 1964

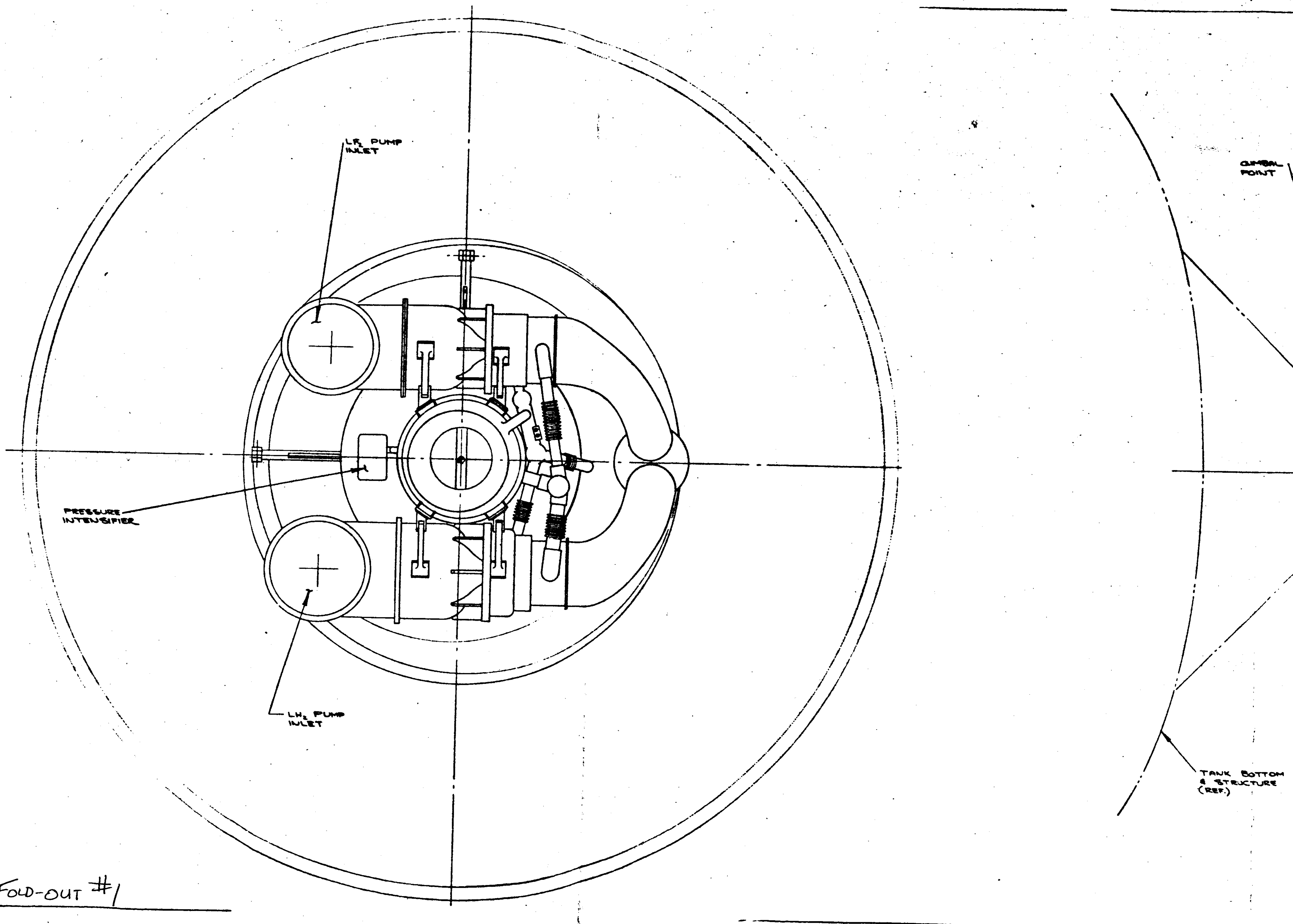
AP-2262

Fold-out #2

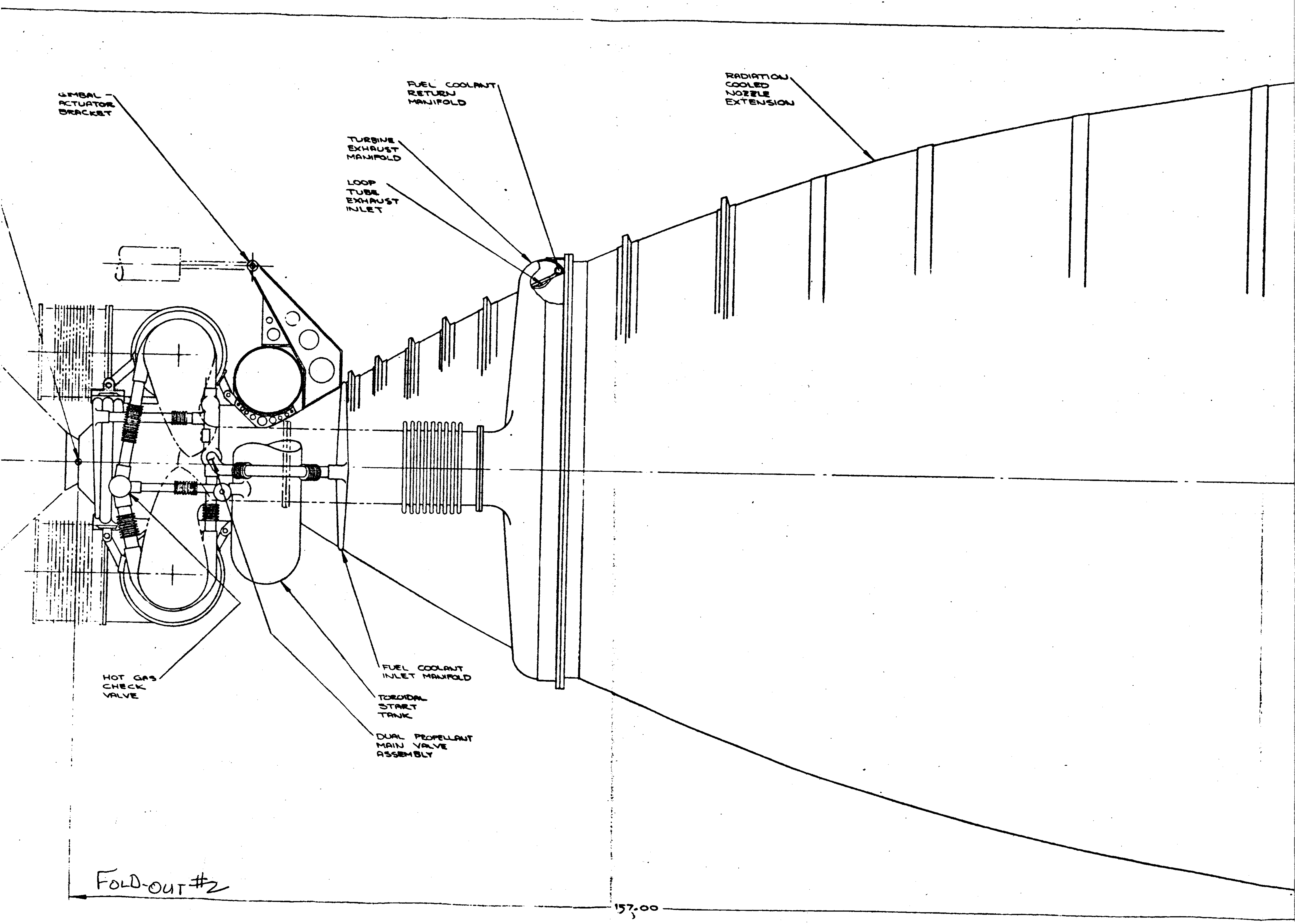
The gimbal point may be positioned as desired by hydraulically altering the point of intersection of the struts. The struts may be compression members facing forward as shown or tension members facing rearward. In the system shown, the propellant inlet lines are integrated into the gimbal struts. This allows the engine to be gimbaled about its C.G. more readily (with less complication of inlet ducting), thus reducing actuator loads. The integrated gas-generator start system is discussed further in the section on subsystems and components.

This concept should be applicable to all the propellants considered in this program. For the cryogenic systems, the increase in heat-transfer associated with the use of multiple inlet lines could make the concept less desirable. There is no inherent thrust-level limit on the use of this concept. However, its applicability to large booster engines will be greatly influenced by the particular propulsion-system package, especially with regard to the use of advanced nozzles.

Figure 50 contains a system layout for a 10K F_2/H_2 engine based on system 201 (Fig. 8). The over-all package is representative of other dual-shaft turbopump systems such as O_2/H_2 systems. In this configuration the turbopumps are mounted horizontally. The turbine-exhaust lines from each turbine meet and flare into one line which dumps into a manifold at an area ratio of 60. Turbine-exhaust gases are used to cool the nozzle beyond


LF2 PUMP INLET
PRESSURE INTENSIFIER
LH2 PUMP INLET
GIMBAL POINT
TANK BOTTOM & STRUCTURE (REF.)

FOLD-OUT #1


GIMBAL ACTUATOR BRACKET
FUEL COOLANT RETURN MANIFOLD
TURBINE EXHAUST MANIFOLD
LOOP TUBE EXHAUST INLET
RADIATION COOLED NOZZLE EXTENSION
HOT GAS CHECK VALVE
FUEL COOLANT INLET MANIFOLD
TOROIDAL START TANK
DUAL PROPELLANT MAIN VALVE ASSEMBLY
157.00

FOLD-OUT #2

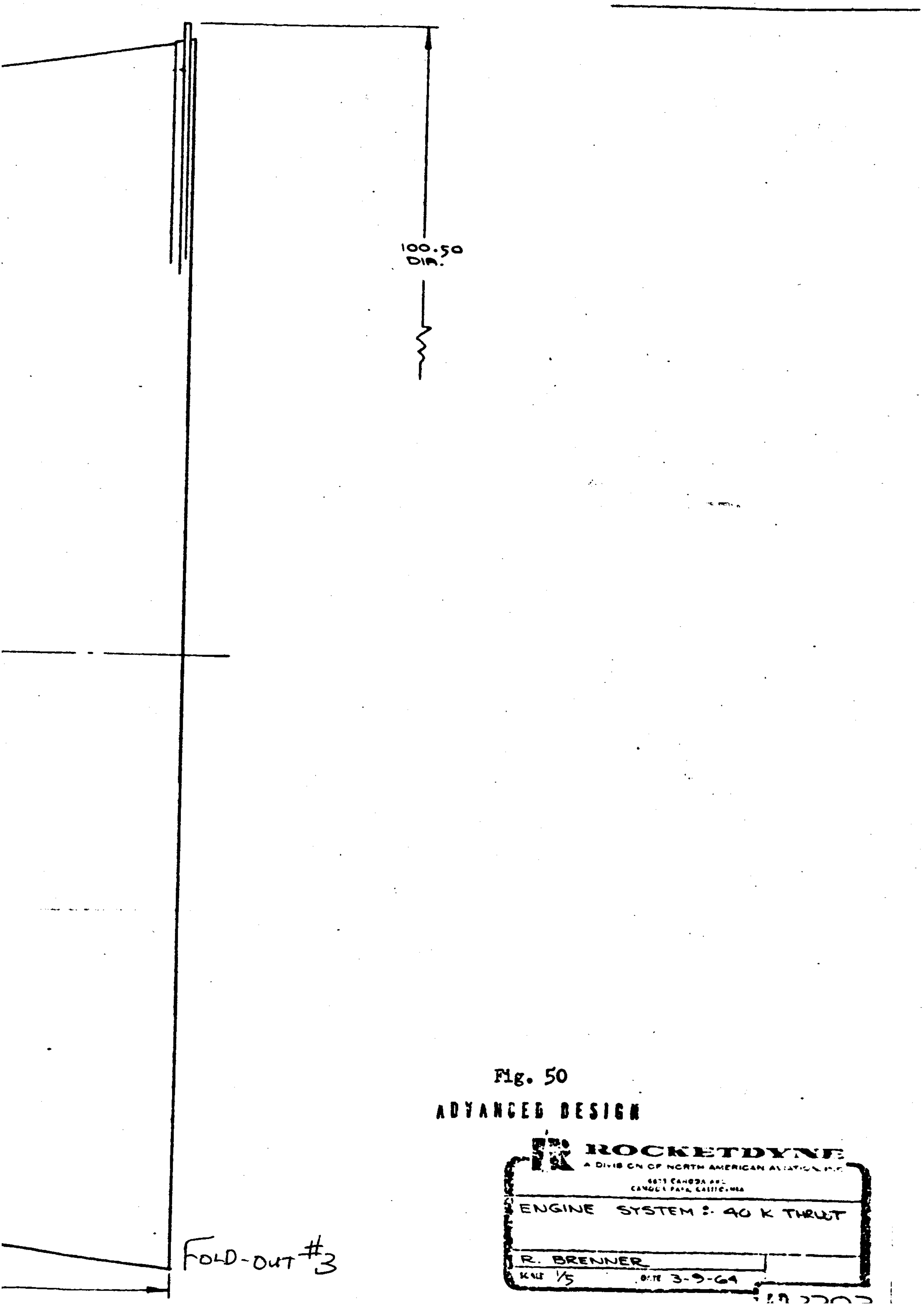


Fig. 50

ADVANCED DESIGN

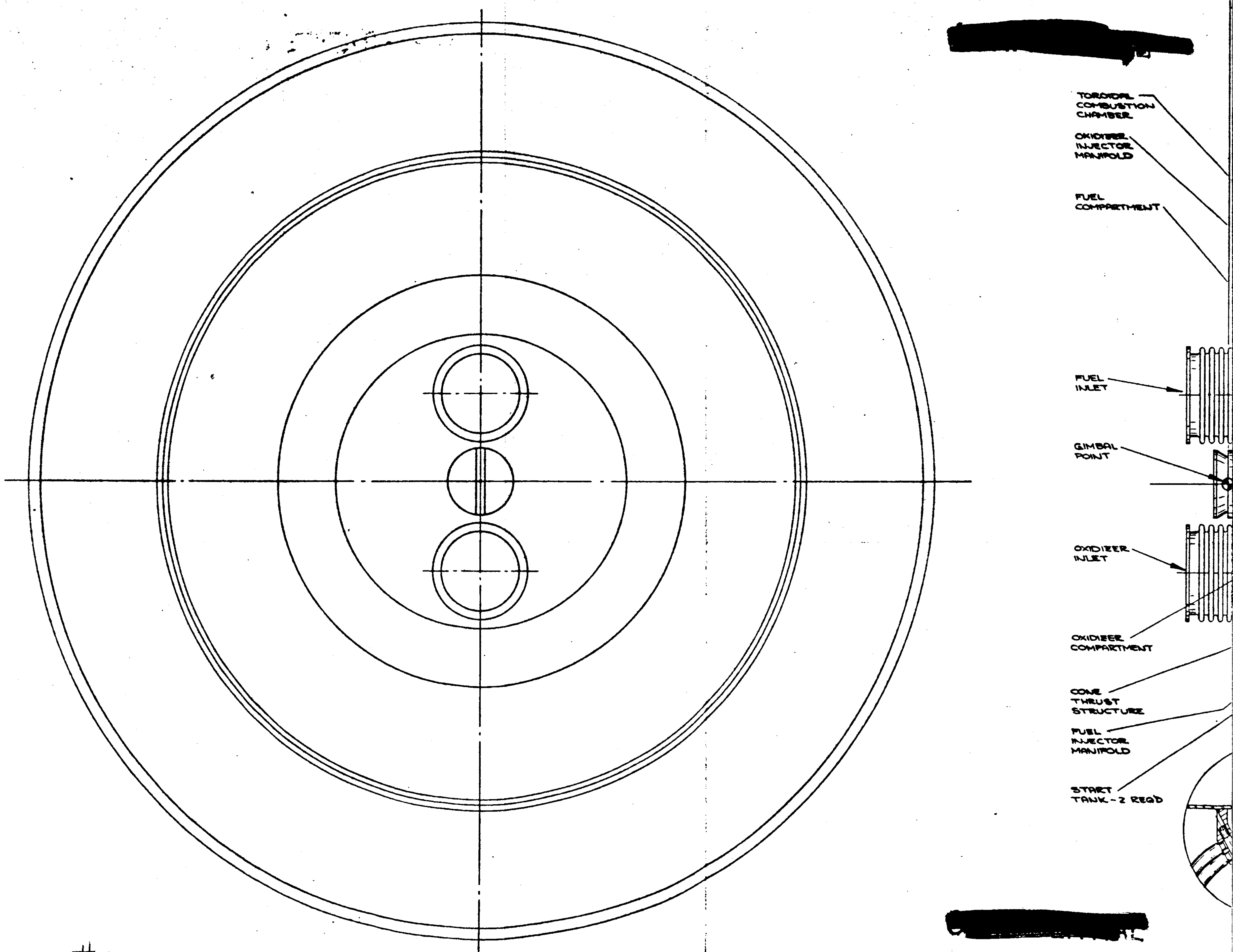
an area ratio of 60; the remainder of the chamber is regeneratively cooled with a 1-1/2-pass system. The start tank (gaseous hydrogen spin-start) is a toroidal tank mounted on the thrust chamber at the throat; it forms an integral part of the structural reinforcement for the throat. Integration of the sizable start-tank in this manner effects a significant reduction in package size. This advantage must be weighed against the associated fabrication problems.

Advanced Nozzles. Figures 51 through 55 contain layouts for spacecraft systems using advanced nozzles. All these systems are shown with annular nozzles, and toroidal combustors (Ref. 11).

Figure 51 is a layout of a LOX O_2/H_2 engine based on system 101 (Fig. 3). The pumps, cylindrical start-tanks, and all the associated components are packaged within the center of the nozzle, and are enclosed in a porous shroud which is cooled by the turbine-exhaust gases.

The fuel and oxidizer are routed to the pump inlets through a cylindrical manifold which is welded to the annular injector. This manifold is divided into a fuel and an oxidizer compartment by a longitudinal member.

[redacted]



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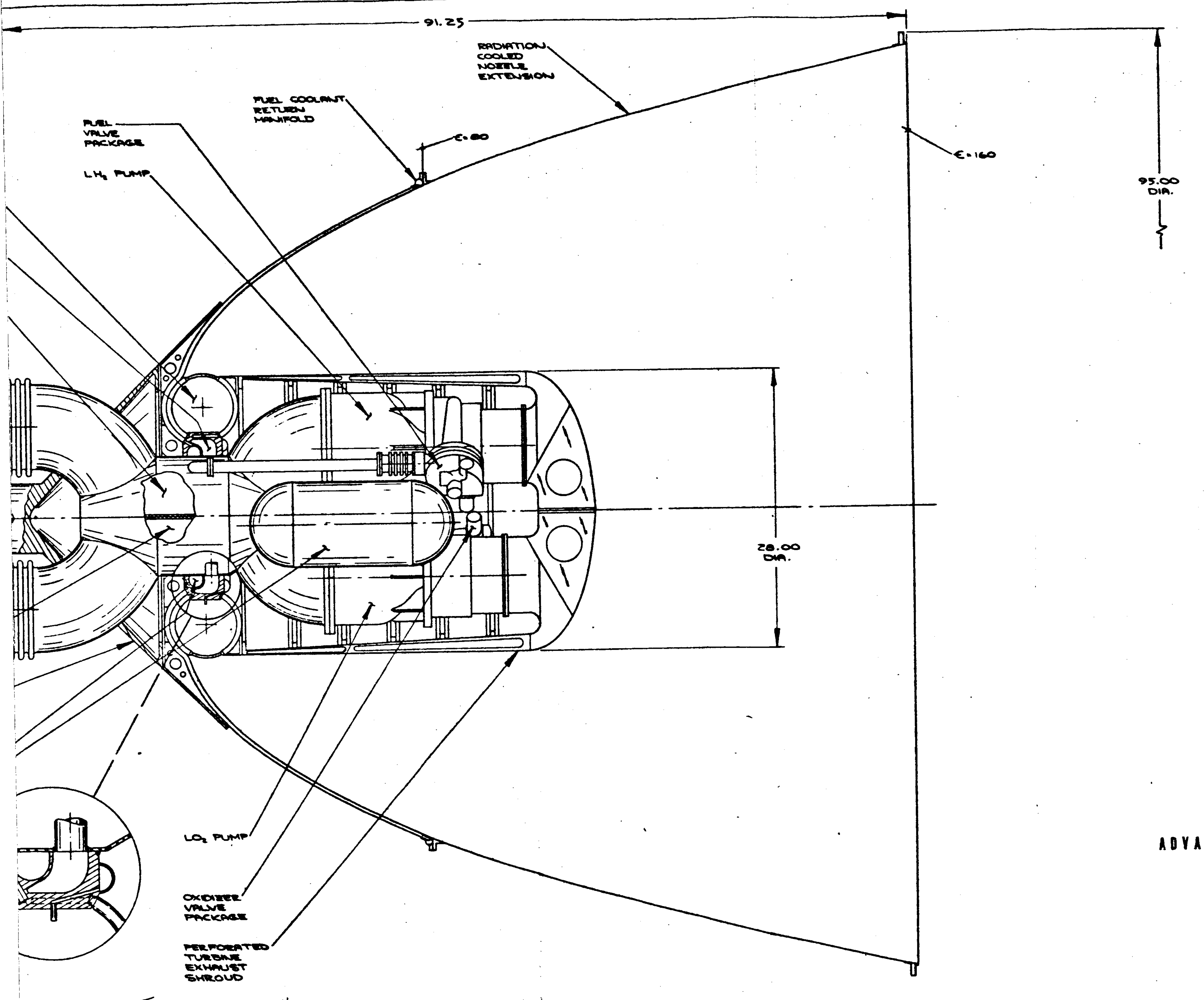
FOLD-OUT #1



FOLD-OUT #2

Fig. 51
ADVANCED DESIGN

ROCKETDYNE
A DIVISION OF NORTH AMERICAN AVIATION, INC.
6633 CANOGA AVE. CANOGA PARK, CALIFORNIA

ENGINE SYSTEM :- 40 K THRUST, LO_2/LH_2, TOROIDAL H-F

HOCHE JR. | SCALE 1/5 | DATE 4-24-64 | AP-2358

The fuel is discharged from the pump, directed through the valve package and enters the injector manifold through the high-pressure line running through the fuel compartment.

Figures 52 and 53 depict a possible configuration for the combined pressure-fed/pump-fed F_2/H_2 system discussed under Analytical Investigations. This design utilizes a centrifugal pump as a pressure source for the fuel, and the oxidizer is pressure-fed by pressurizing the main oxidizer tank with stored gas.

The fuel-pump volute is integrated into an annular manifold, which envelopes the pump. The manifold receives the fuel from the pump, directs it through a double-walled combustion-chamber (as a coolant), and supplies the annular injector with the pump-fed propellant. The pressurized oxidizer is supplied to the annular injector through multiple poppet valves radially oriented and integrated into the injector.

The turbine tap-off is taken from the combustion chamber in four equally spaced places, and subsequently exhausts its gases into the nozzle core.

Use of this concept should probably be limited to spacecraft applications, since the maximum practical chamber pressure is not sufficiently high for most booster applications. Its use is most attractive for an easily-developed, comparatively-inexpensive F_2/H_2 system; the ease of development and low cost result from no fluorine pump being required.

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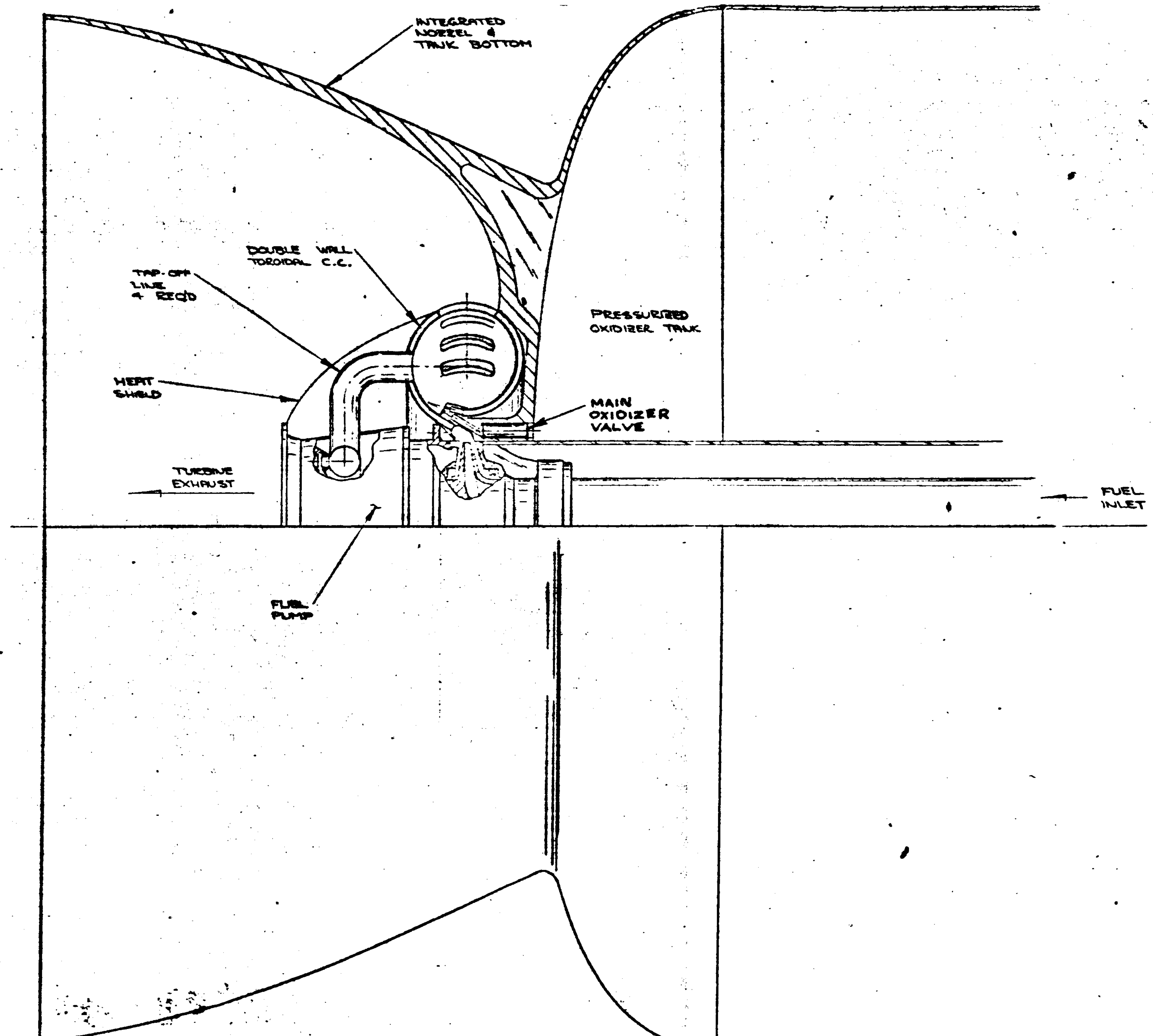


[redacted]

Fig. 52

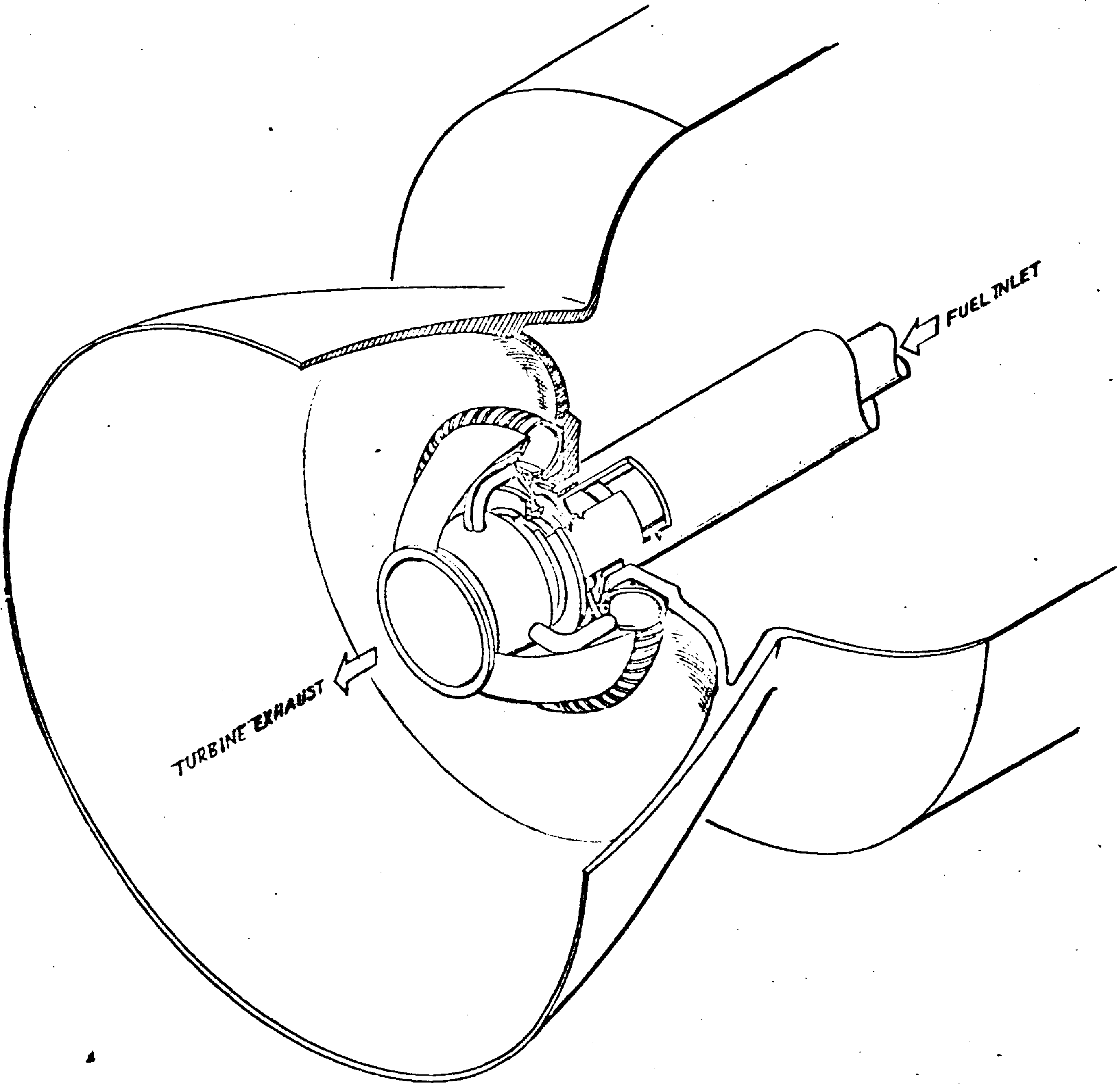
ADVANCED DESIGN

ROCKETDYNE
A DIVISION OF NORTH AMERICAN AVIATION INC.
CANOGA PARK, CALIFORNIA

INTEGRATED ENGINE CONCEPT 2- COMBINED PUMP & TANK PRESSURE SYSTEM

H. A. HOCHE	JR
SCALE NONE	11-20-63

AP-2231



123

Fig. 53 Combined Pump-fed and Pressure-fed (Perspective)

Figures 54 and 55 show two system configurations for an NTO/50-50 system. These configurations feature complete integration of system controls and valves into the thrust structure. The fuel-inlet duct in Figure 54 uses a standard bellows. This requires (for gimbaling) that the duct have an elbow between two bellows which is structurally undesirable. The configuration shown in Figure 55 uses the canted bellows to improve the structural situation; use of the canted bellows eliminates having a heavy elbow supported by bellows on both ends.

Boosters. Preliminary-design layouts of representative booster configurations for dual-shaft and single-shaft turbopump systems are presented in Figures 56 and 57 . Both systems use the aerodynamic-spike nozzle-concept (Ref. 11).

The system in Figure 56 is an O_2/H_2 system, and is based on the system 103 configuration (Fig. 5). Four sets of turbopumps (2 per set) were used for this design. However, this should not be interpreted as indicating that this is necessarily an optimum number; it was selected as being representative of multiple-pump systems.

The four LO_2 pumps are attached directly to the bottom of the LO_2 tank and their turbine exhaust dumps directly into the secondary-flow region. The four LH_2 pumps are fed from a single line which passes through the center of the LO_2 tank. Their turbine exhaust also dumps directly into the secondary flow region. This arrangement minimizes inlet and exhaust ducting and provides a simple and efficient support

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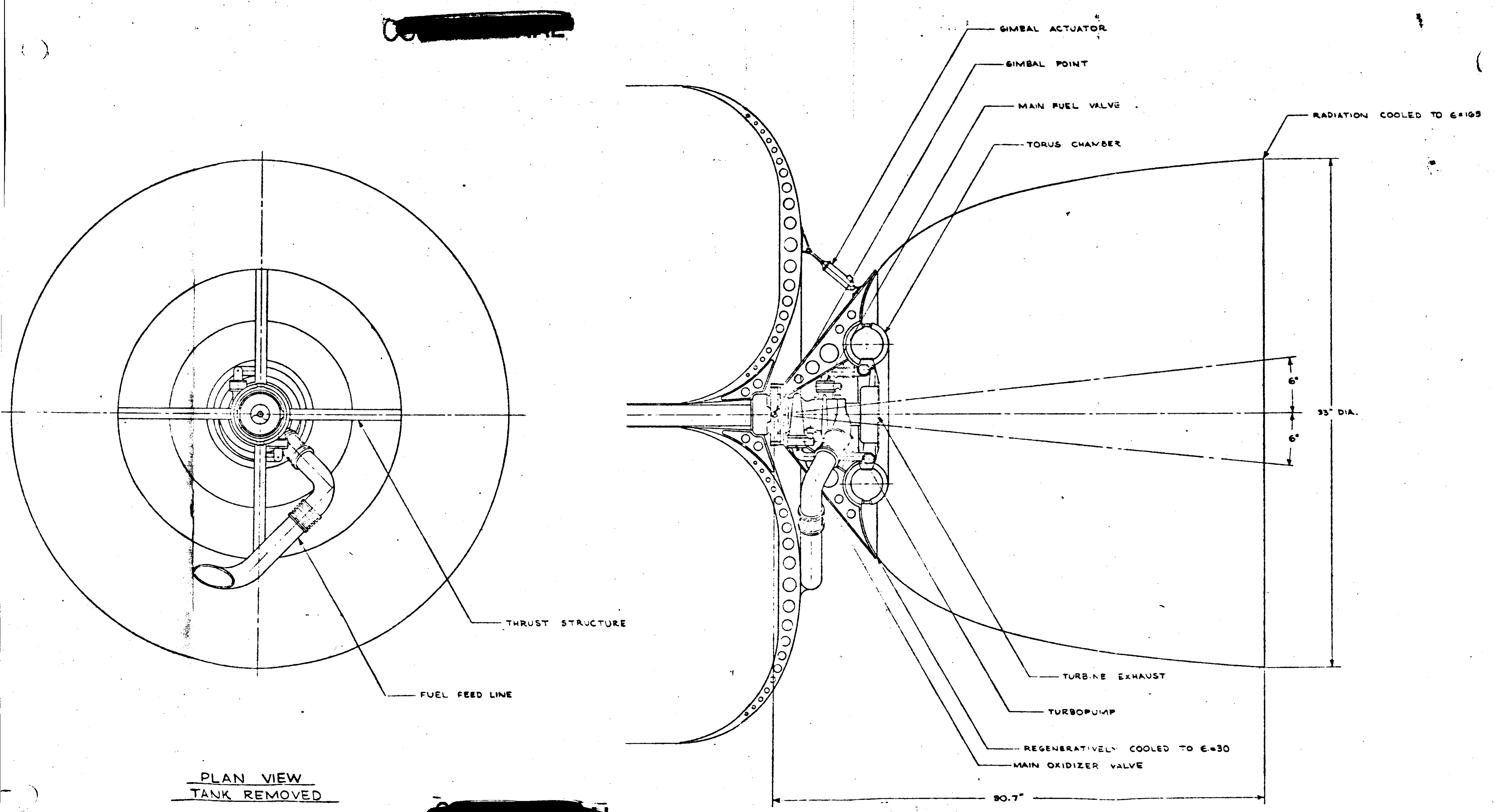

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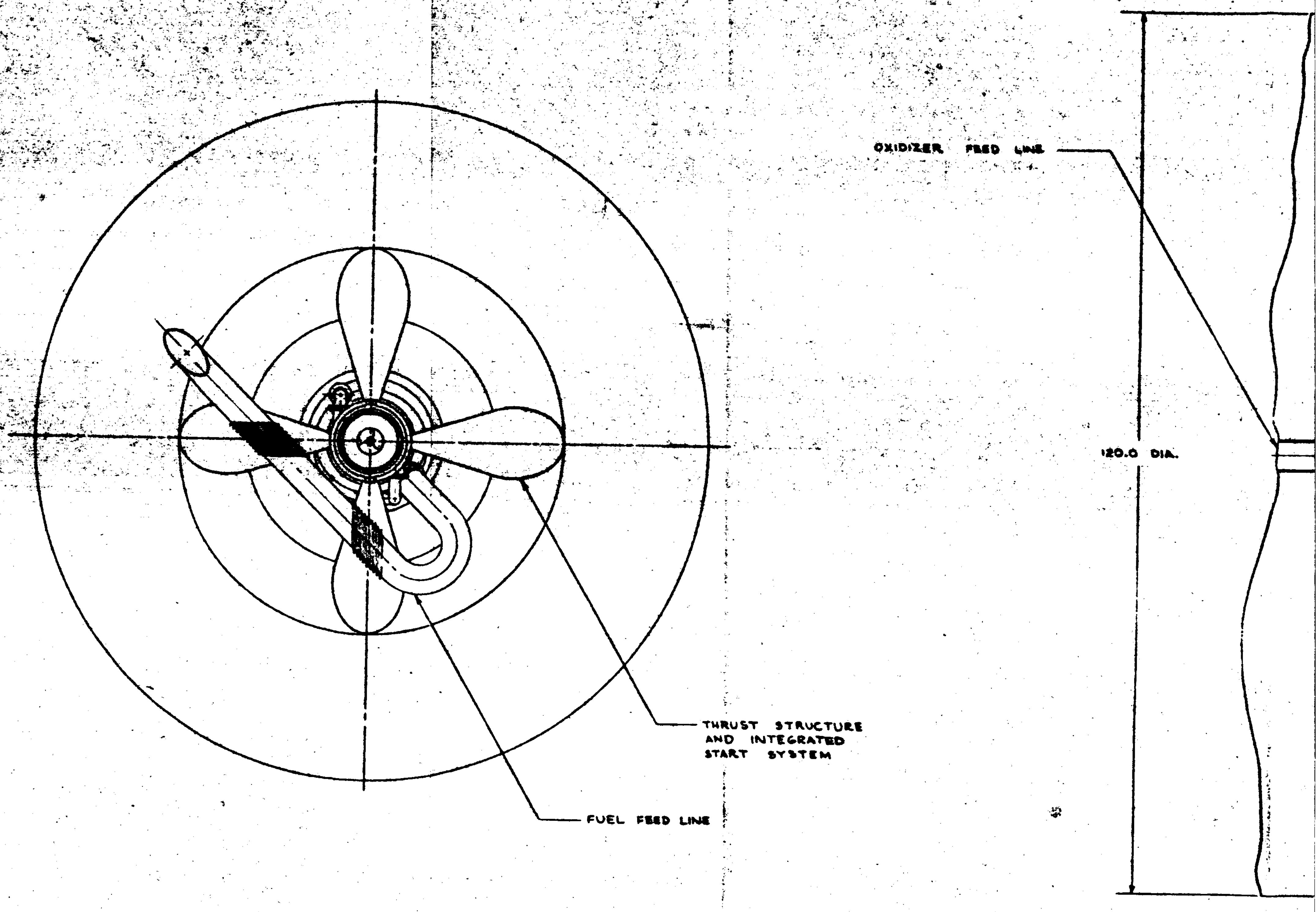
Storable 40K E-D Engine
Fig. 54



PLAN VIEW
TANK REMOVED

FOLD - OUT #1

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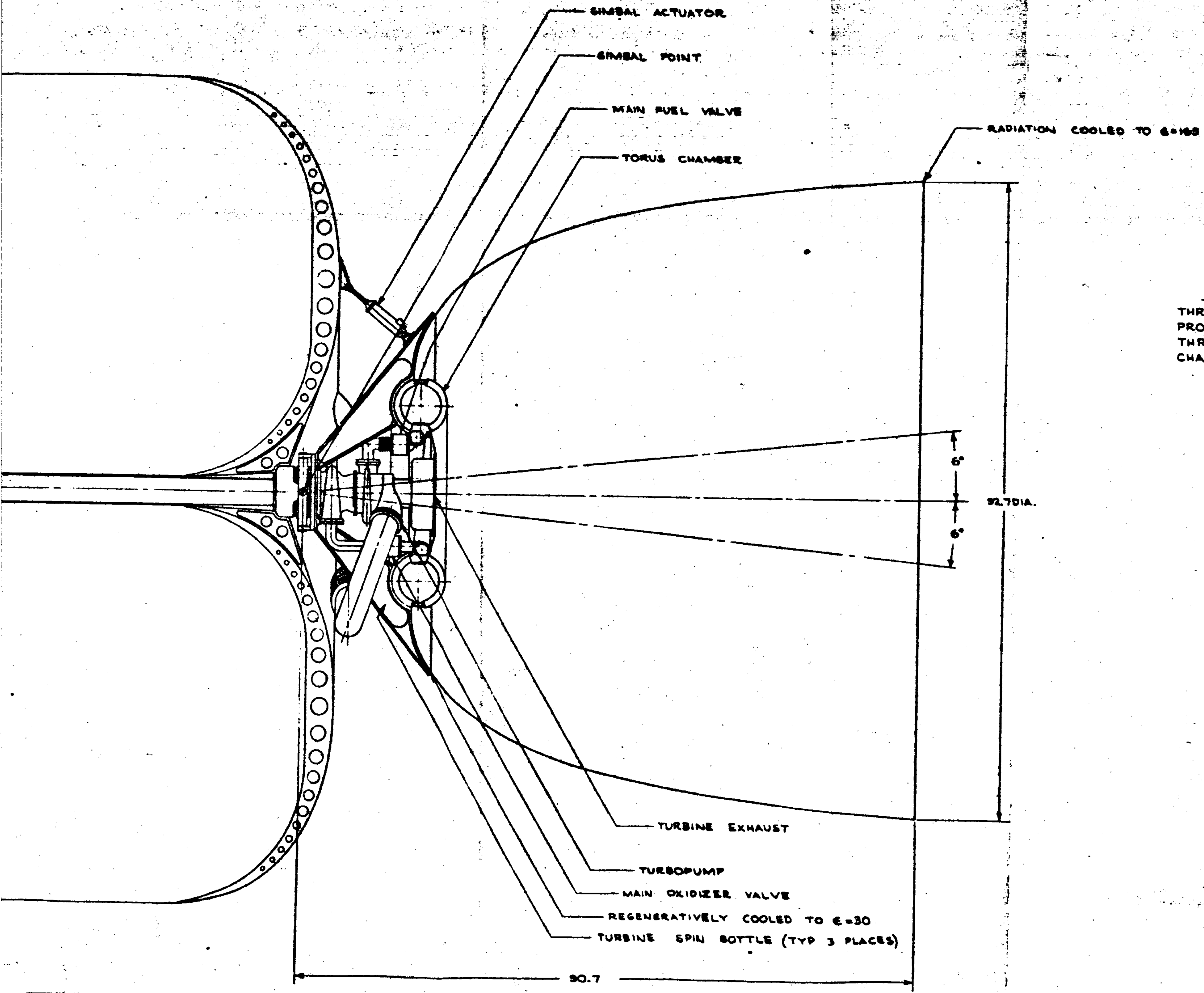


PARAMETERS

THRUST	40 K
PROPELLANTS	NTO/50-50
THRUST CHAMBER ϵ	165
CHAMBER PRESSURE	500 PSI

Fig. 55

E-D ENGINE

By: R. K. Russell 4-13-64

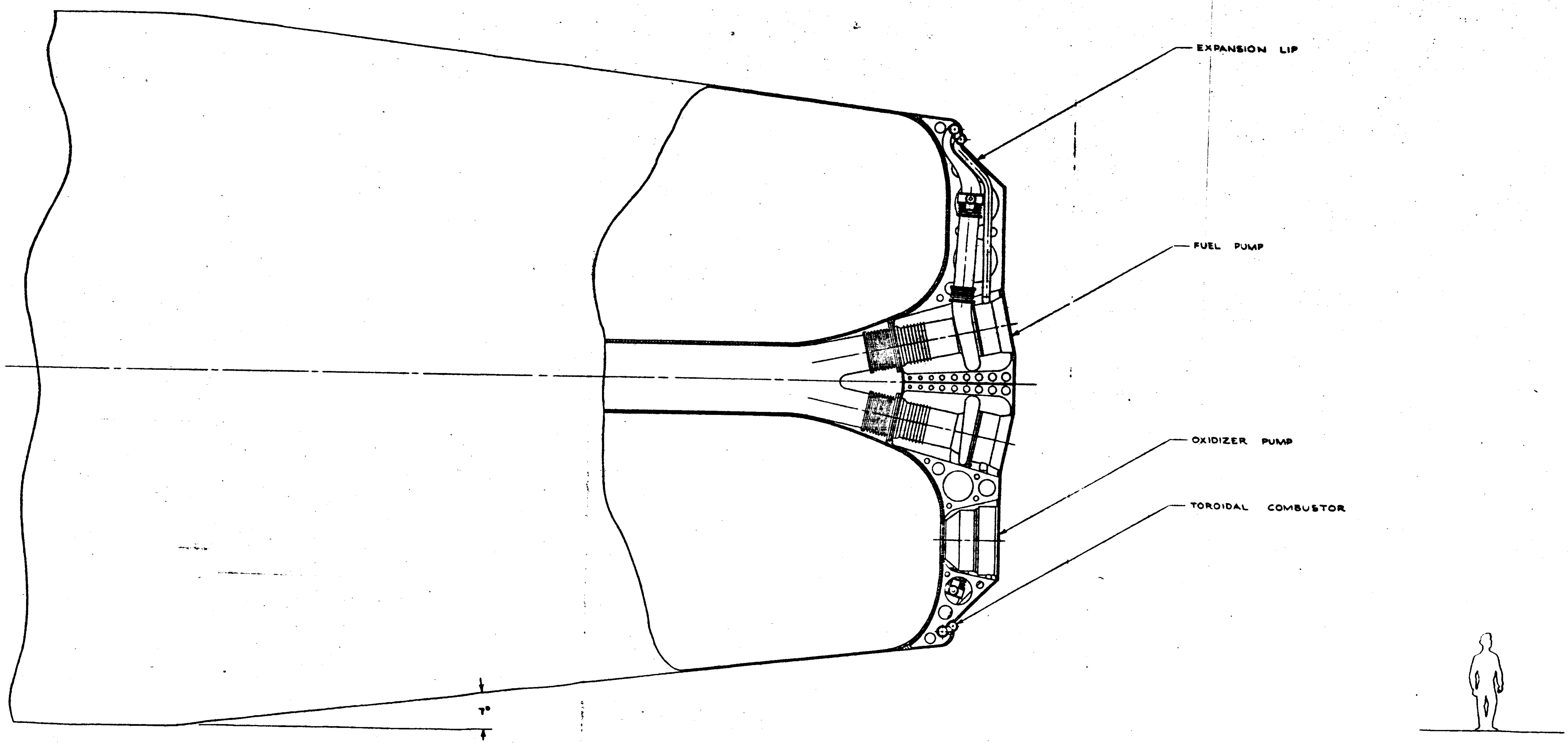
1/8

[redacted]

FOLD-OUT #2

2339

[redacted]



SECTION A-A

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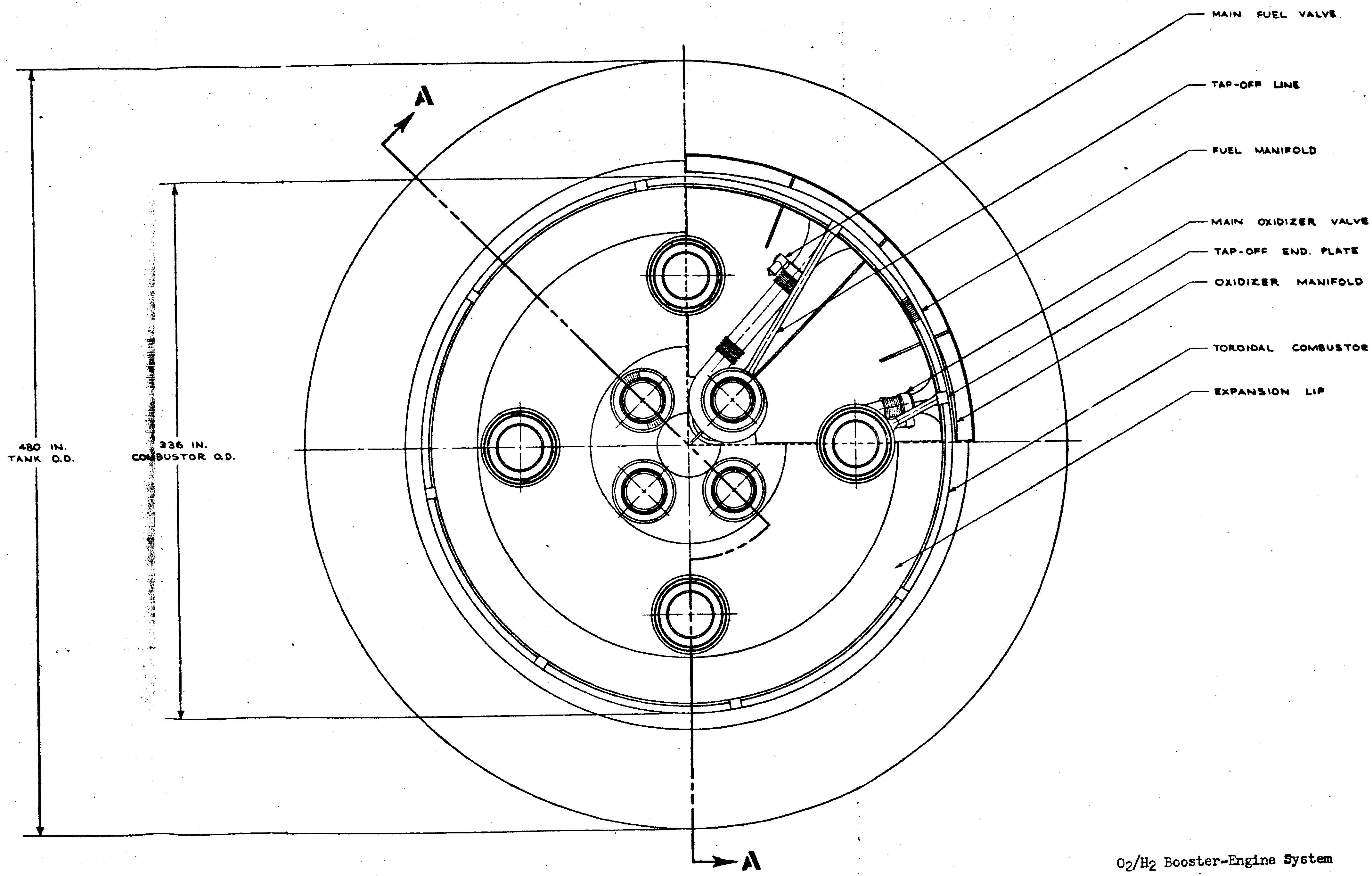
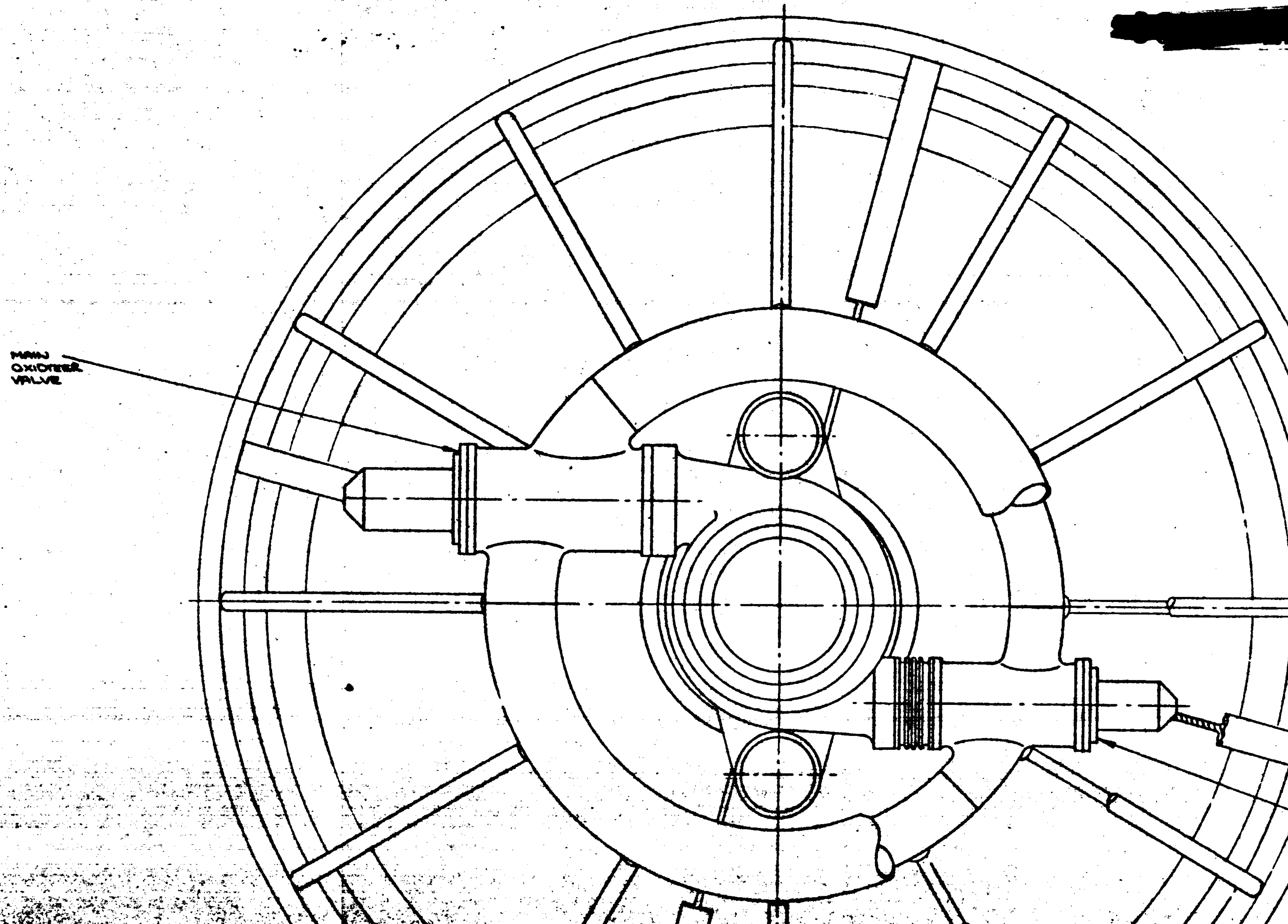
FOLD-OUT #1



O_2/H_2 Booster-Engine System

Fig. 56

FOLD-OUT #2



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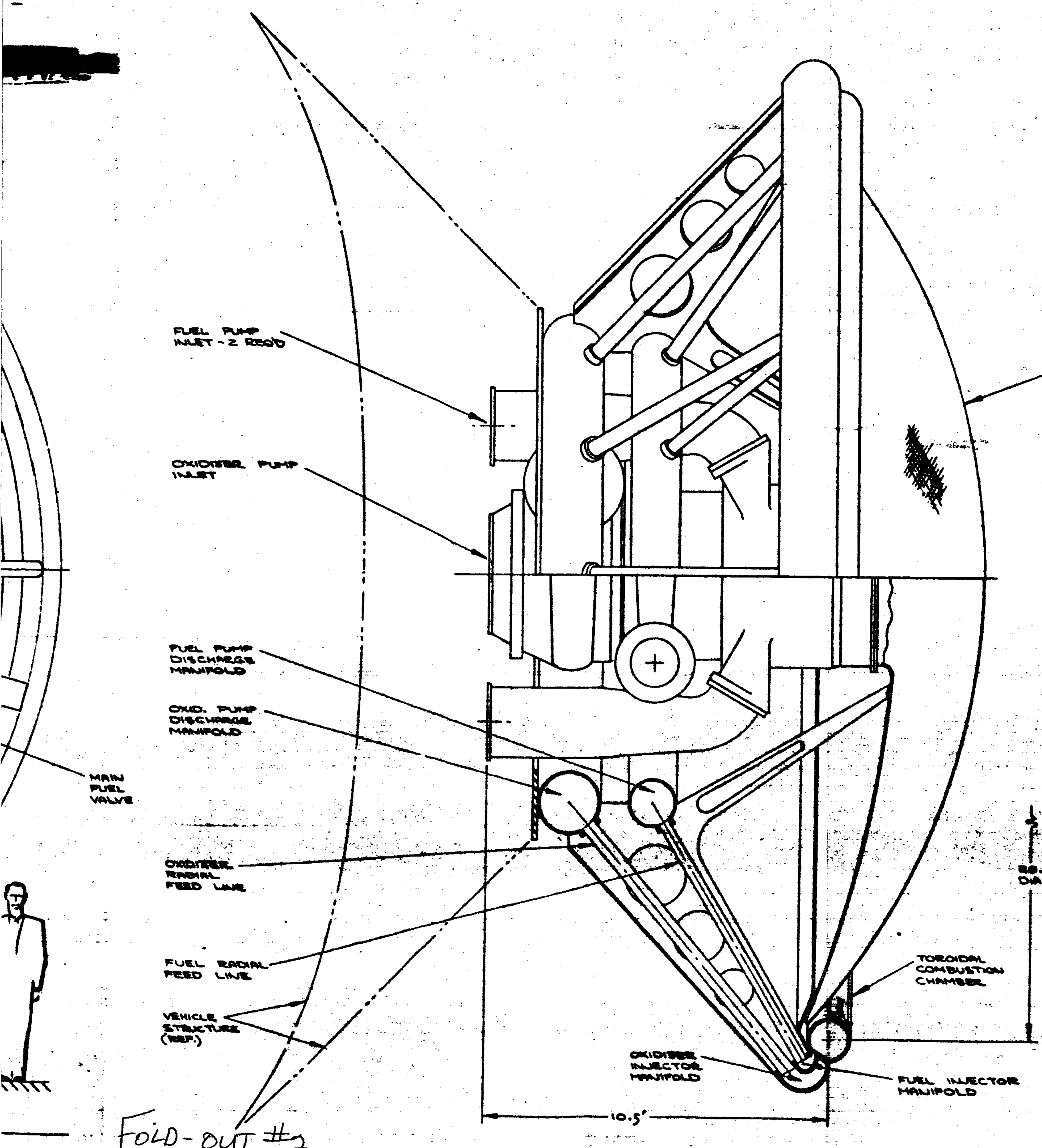

Fig. 57

ADVANCED DESIGN

ROCKETDYNE

AEROSPIKE ENGINE SYSTEM 2-6 M THRUST, LO_2/RP1, SINGLE PUMP ASSY.

HOCHE JR.

SCALE 1/20 DATE 3-3-64

AP-2286

for the pumps. Pre-valves for the LO_2 pumps may be integrated into the pump inlets. For the LH_2 pumps, the pre-valves may be integrated into the bottom of the LH_2 tanks, using one valve for all four pumps. A separate main valve and tapoff line is provided for each pump.

The tapoff gas is removed from the chamber at the end plates (Fig. 58). The need for a collector manifold along the length of the chamber and tapoff ports through the tubes or main injector is thereby eliminated. An injector is built into the end plate (to reduce the local mixture ratio); it may be fed from the main chamber manifolds. The propellants injected into the end plates could comprise most of the tapoff flow. This could minimize side flow from the main chamber into the end plates.

Figure 57 contains a preliminary-design layout of an O_2/PR-1 six-million pound thrust engine which is based on system 401 (Fig. 20). This design uses a single turbopump, which again, is not necessarily optimum; it was chosen to aid in considering concepts associated with single as well as multiple turbopump systems. This configuration integrates: (1) the pump-discharge manifolds and thrust structure, (2) the hypergol cartridge and injector manifolds, and (3) turbine-drive power-source and main combustor (tap-off concept). A comparison of the sizes of the standard six-foot man shown in Figure 57 and the main propellant valves underscores the packaging advantage associated with the multiple-poppet valve concept (see Components section).

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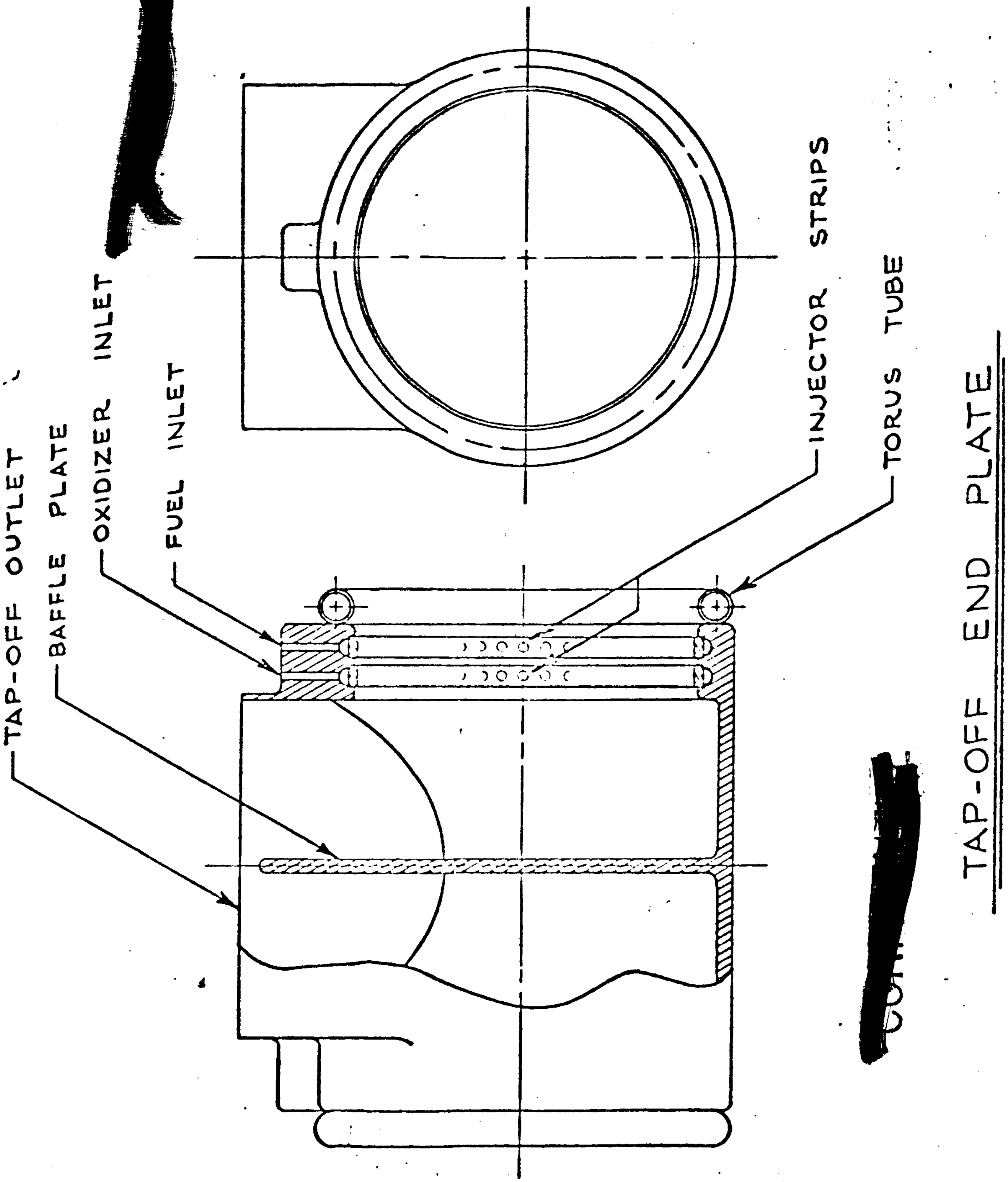

[redacted]

Fig. 58

In Figure 57, propellants are discharged into toroidal manifolds which encircle the turbopump. These manifolds distribute the propellants to multiple radial feed-lines that tie into the annular injector-assembly. These manifolds also act as structural support for the turbopump, and carry the thrust loads from the thrust chamber. The main-propellant-valve housings are integrated into these manifolds. The main-propellant valves are poppet-type valves.

The secondary flow required for the aerospike is otbained from the turbine exhaust. This exhaust is discharged through the perforated centerbody. This secondary flow also acts as a coolant for the centerbody structure.

Figure 59 shows some details of the toroidal combustion-chamber and coaxial injector. Fuel is discharged into the "fuel injector manifold" and flows through the combustion chamber tube as a coolant. Similarly, the oxidizer is discharged into the "oxidizer injector manifold" and flows directly through the coaxial injector. In Fig. 60, the hypergol cartridge is shown mounted on the annular injector. The igniter-fluid inlet-passage is integral with the annular injector. The hypergol cartridge can readily be replaced. Any number of these cartridges can be used on an engine system to increase reliability.

[redacted]

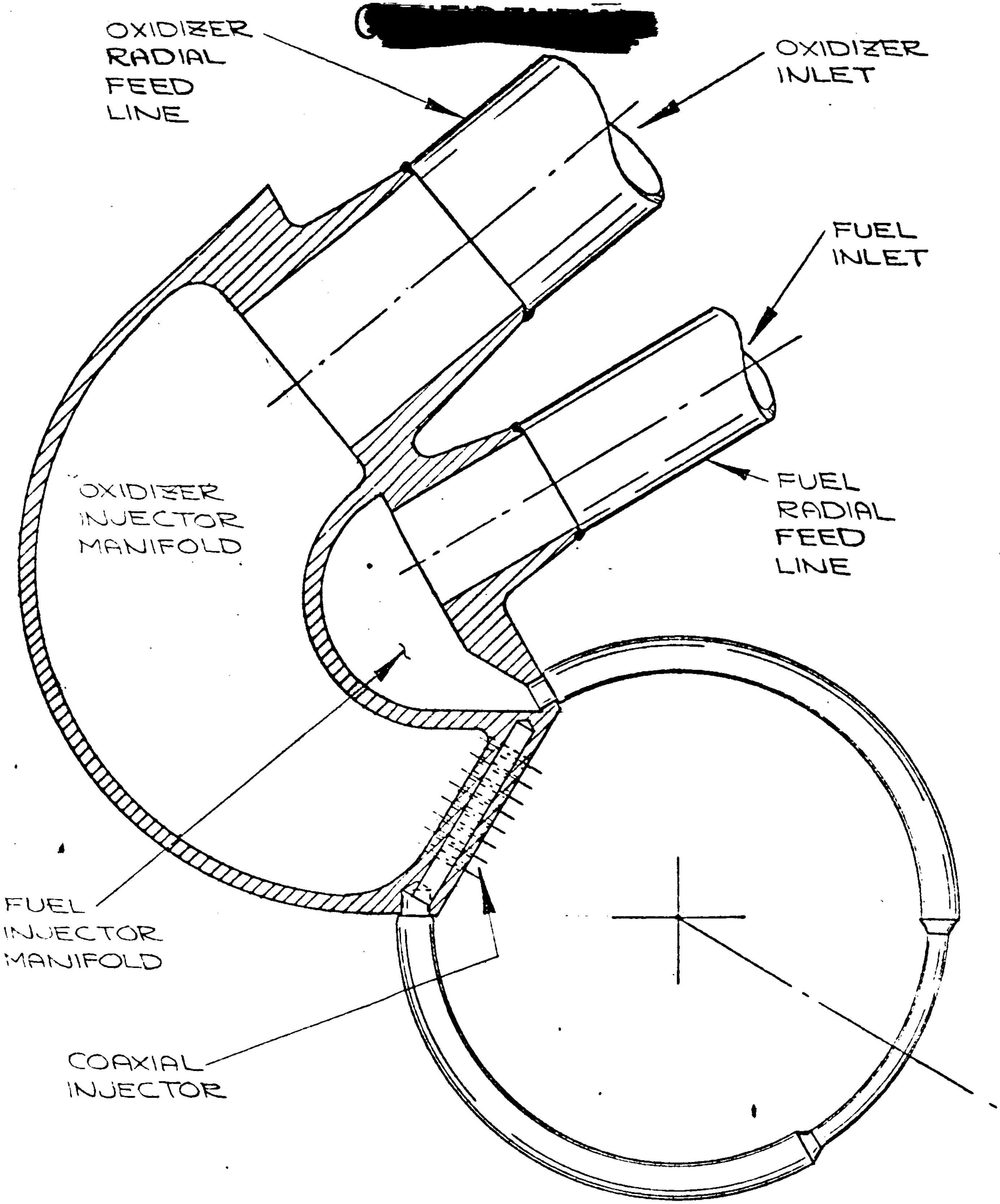

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Fig. 59. Injector Manifold Detail

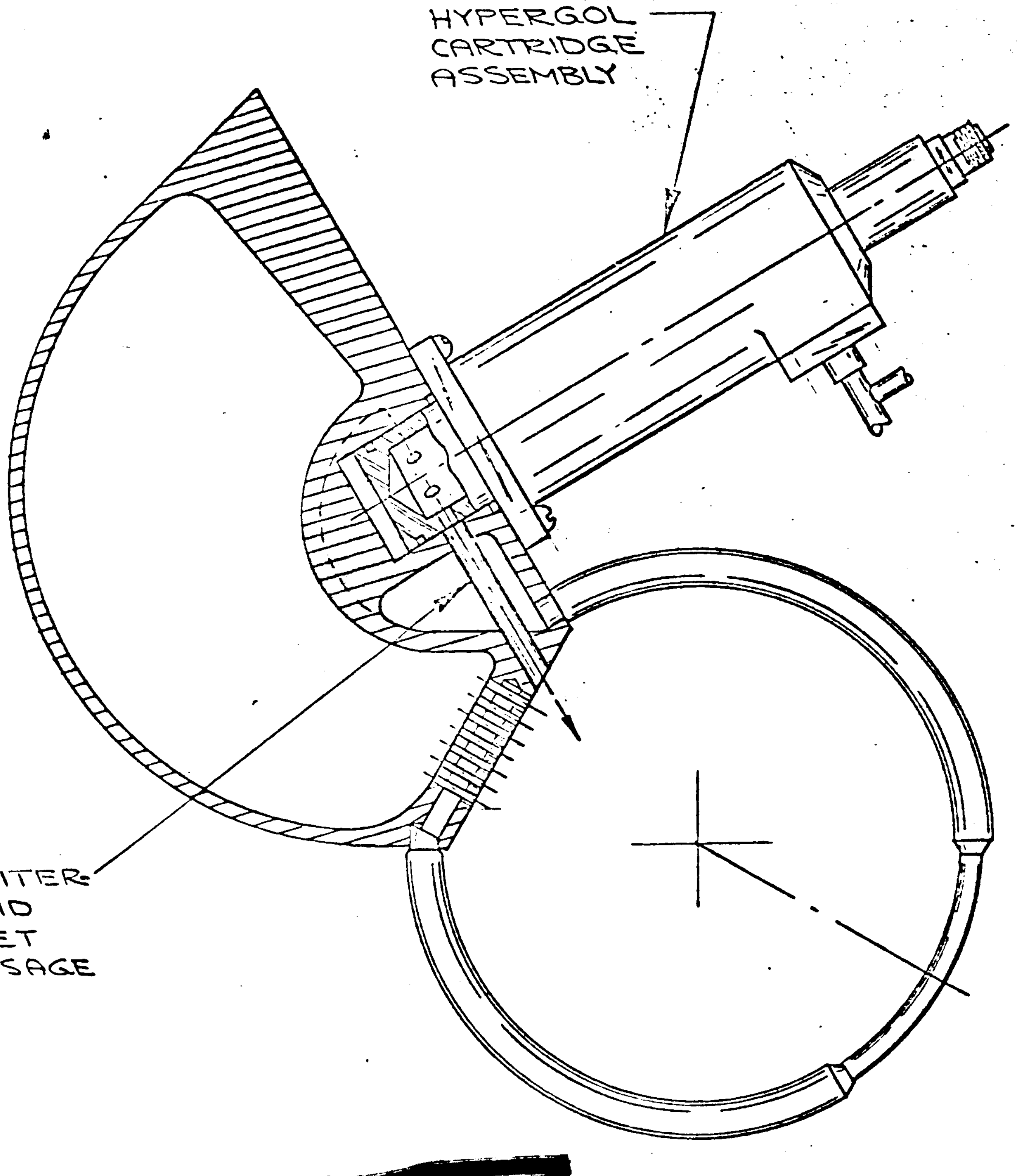


Figure 60

Hypergol Cartridge Installation

Components and Subsystems

In the paragraphs that follow, the various component and subsystem concepts evolved and considered during this program are described and discussed. Some of these concepts involve integration of specific components of the selected basic systems, while others have more general applicability.

Propellant-Actuated Valves. Figures 61 through 66 contain preliminary design layouts of integrated propellant-actuated valves.

The valves shown in Figure 61 and 62 were designed for use in a 40K NTO/50-50 defined by the schematic for system 307 (Fig. 19).

Figure 61 depicts the main control valve (MCV) which is an integrated valve-package containing the main fuel-valve, the oxidizer igniter-valve, and a control solenoid valve. The operation of this valve can best be understood by referring to the operational sequence for system 301 (similar to system 307) in the section on concept definition, and following the valve-operation schematic on Figure 61 . This is essentially an in-line poppet valve. The propellant flows around a centerbody, which is supported by three (or more) ribs. The poppet seals in the closed position against a soft metal seat. A combined bellows and spring is utilized both as a dynamic seal and for closing the poppet.

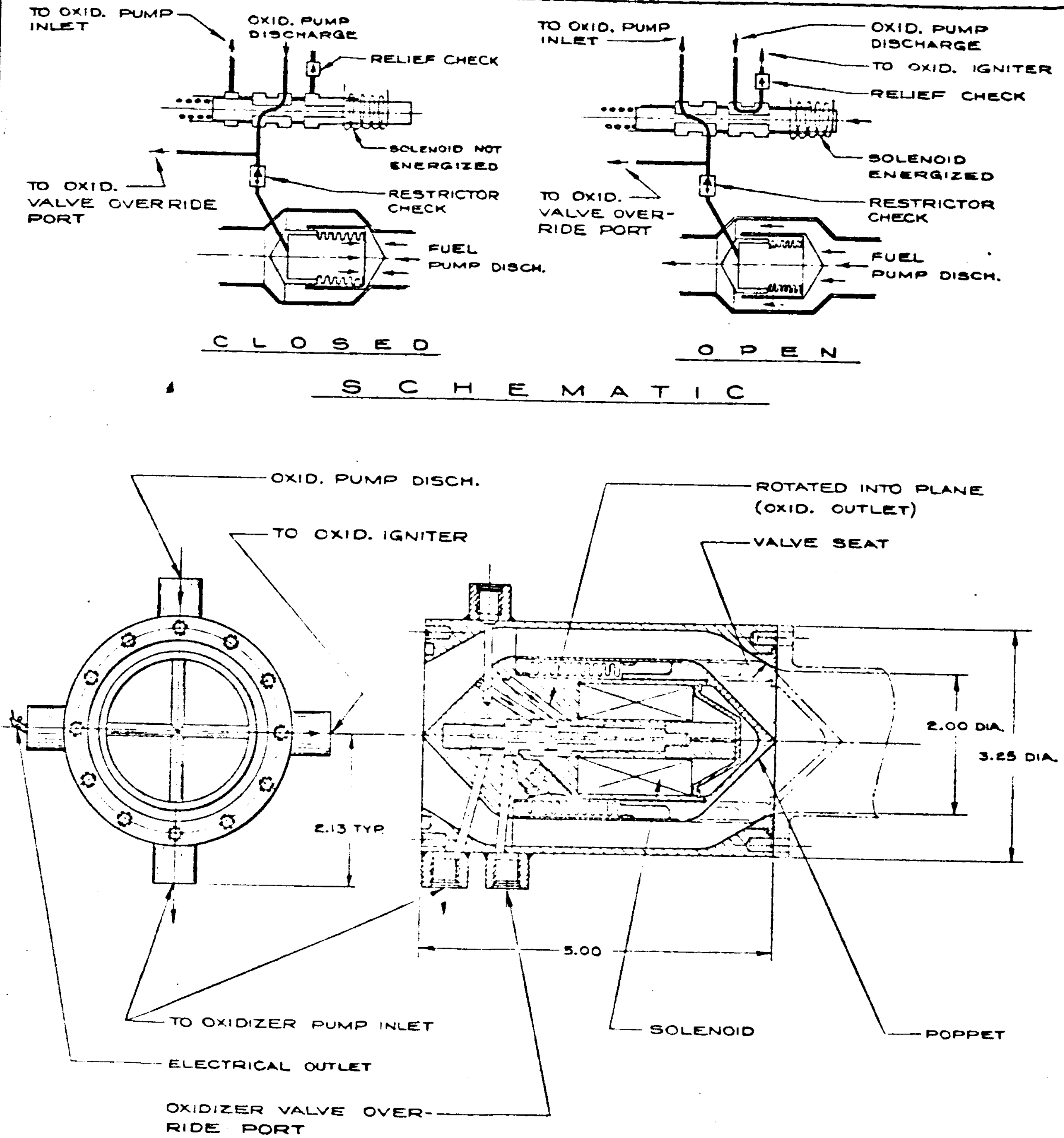


Fig. 61

ADVANCED DESIGN

ROCKETDYNE
A DIVISION OF NORTH AMERICAN AVIATION, INC.
6633 CANOGA AVE. CANOGA PARK, CALIFORNIA

COMBINED FUEL VALVE, OXID. IGNITER VALVE AND ENGINE START SOLENOID

T. E. COWELL

SCALE FULL SIZE | DATE APR. 14, 1964

REV. 5-17-64

AP-2340

The control solenoid is enclosed in the centerbody. Electrical leads are wired through one of the support ribs. The solenoid actuates a spool valve, which is utilized as the oxidizer igniter valve. The igniter flow passes through the support ribs. The vent port is connected to the oxidizer pump inlet to prevent contamination and propellant loss.

Figure 62 contains the oxidizer valve for system 307 (Fig. 19). It is recalled that this valve is actuated by ignition-stage chamber-pressure (sensed at one of the propellant inlet manifolds). The valve is closed by oxidizer-pump discharge-pressure when the control solenoid is de-energized at cutoff. This is shown schematically in Figure 63 .

These valve concepts are not inherently restricted to lower thrust levels. For large-thrust engines, the support-rib size required for passage of the igniter-oxidizer flow may be an important consideration. Use of the main-control-valve concept with cryogenic propellants may not be feasible because of possible freezing resulting from the igniter-oxidizer flow passing through the main fuel-valve.

Figures 64 and 65 contain preliminary design layouts of the main oxidizer and main fuel valves for a 40K O_2/H_2 engine based on the schematic for system 101 (Fig. 3).

The valve shown in Figure 64 is an oxidizer diaphragm-valve, a solenoid

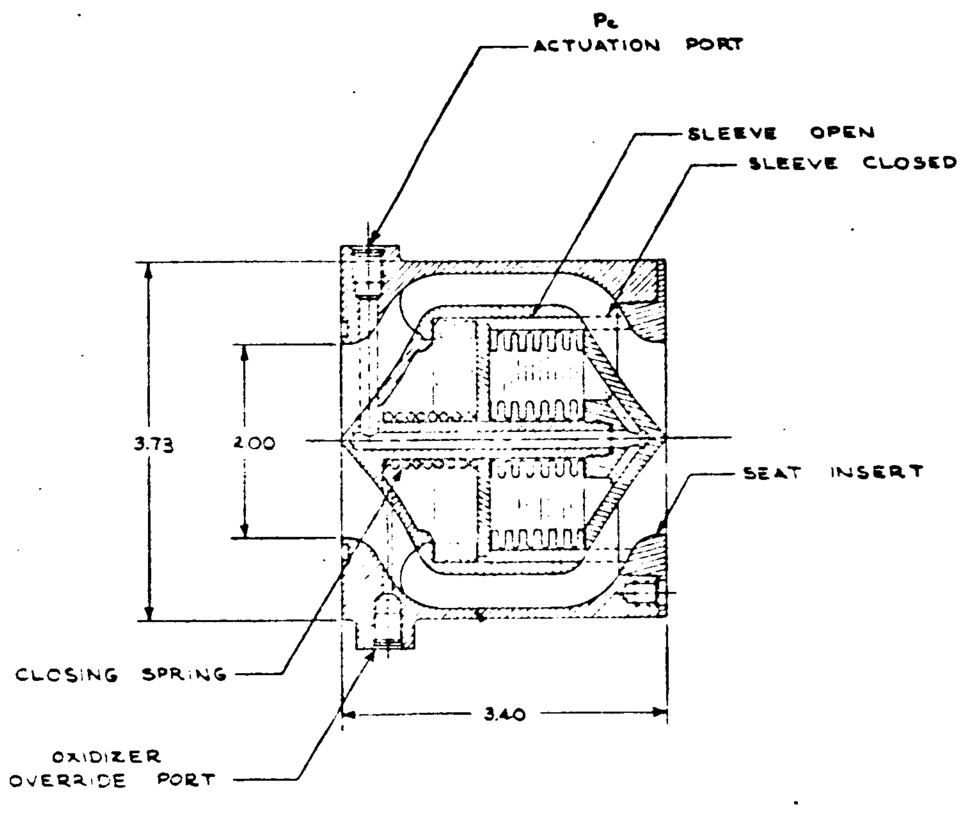


Fig. 62

ADVANCED DESIGN

ROCKETDYNE
A DIVISION OF NORTH AMERICAN AVIATION, INC.
6633 CANOGA AVE.
CANOGA PARK, CALIFORNIA

AXIAL FLOW
SLEEVE VALVE

L.H. Russell

SCALE FULL | DATE 4-10-64

REV. 5-14-64

AP-2336

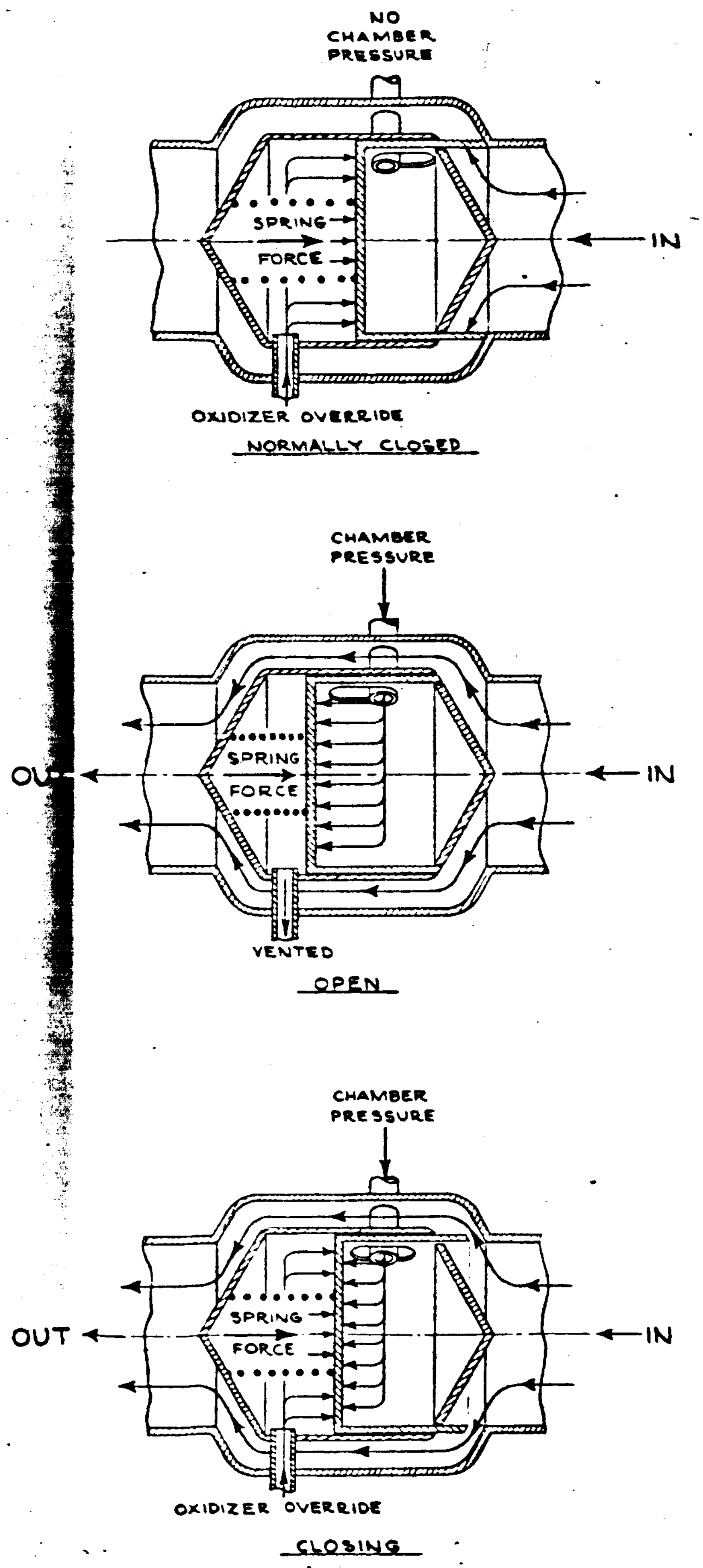


Fig. 63

ADVANCED DESIGN

ROCKETDYNE
A DIVISION OF NORTH AMERICAN AVIATION, INC.
6633 CANOGA AVE.
CANOGA PARK, CALIFORNIA

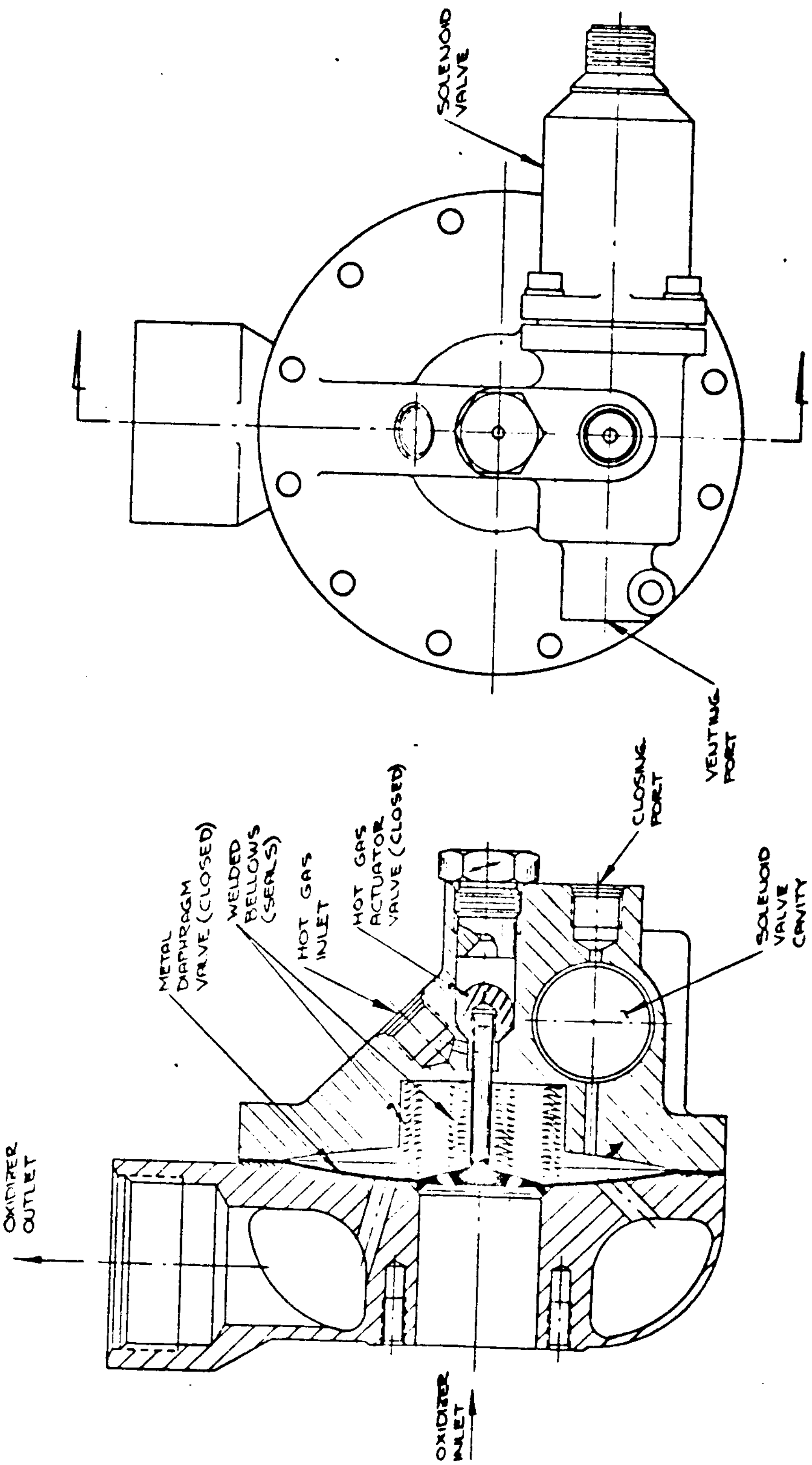
P_c ACTUATED OXIDIZER VALVE SCHEMATIC

L.H. RUSSELL

SCALE NONE | DATE 5-8-64

REV. 5-14-64

AP-2372



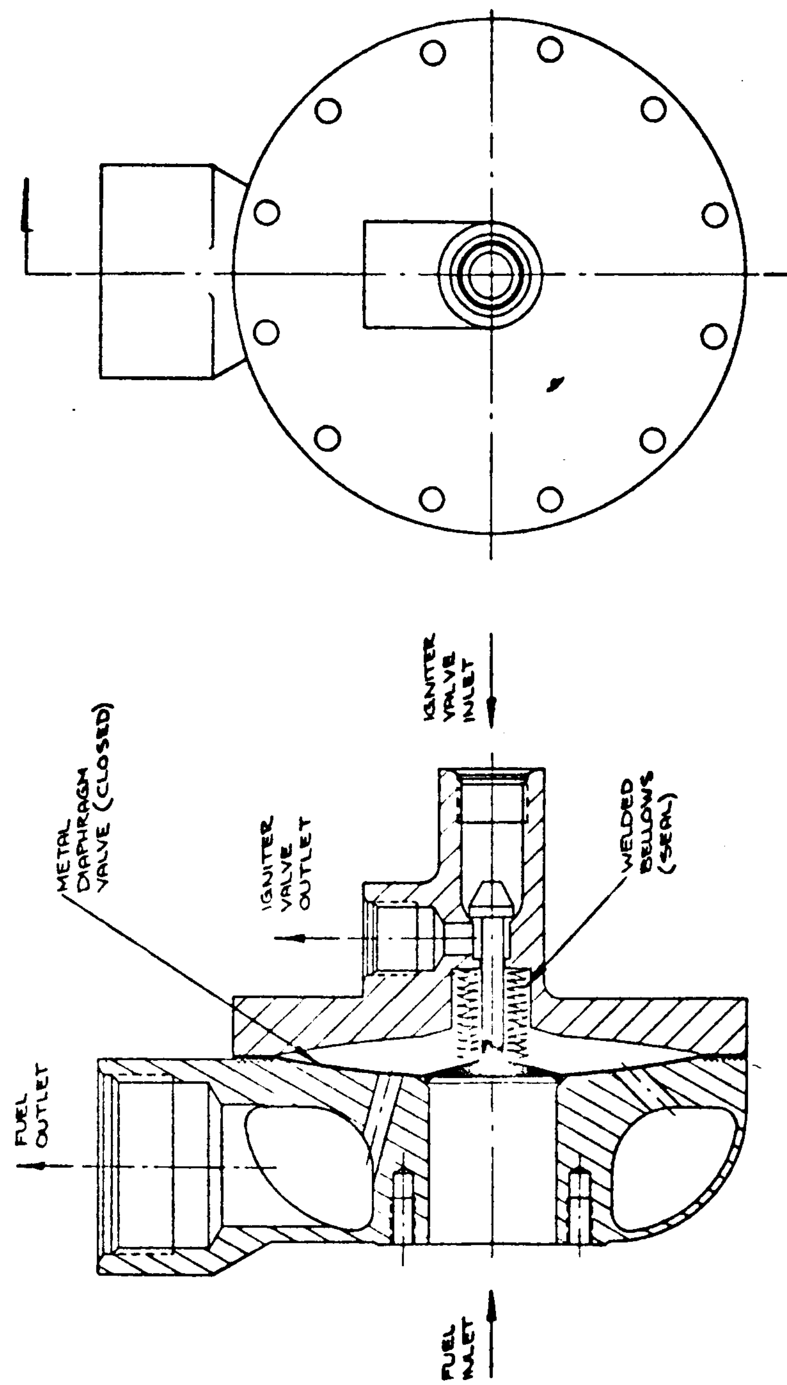
MAIN OXIDIZER VALVE

Fig. 64

control-valve, and a hot-gas actuator valve in a common assembly. This packaging technique eliminates long feed lines, metal to metal connect points and provides a lightweight component assembly.

This design employs a circular metal disc as a diaphragm; it is sealed around the periphery by a series of concentric serrations. To provide a normally closed valve, the circular metal disc is performed so it is spring-loaded closed. Prior to opening, oxidizer pressure is applied to both sides of the metal disc. Because of the area difference, the valve remains closed until ignition-stage chamber-pressure (acting on the hot-gas actuator) overrides the closing force. Valve closure at cutoff is effected by main-line oxidizer-pressure. This pressure is directed to the back side of the metal disc by opening the solenoid control valve. This solenoid valve is a normally open valve that is energized (closed) by the start signal. During system operation, the valve is closed and the back side of the metal disc is vented to the suction side of the oxidizer pump.

The main fuel-valve for the same system is shown in Figure 65 . It is similar in design to the main oxidizer-valve. In this valve, the fuel diaphragm-valve, the solenoid control-valve, and the oxidizer igniter-valve are packaged into a common assembly. The solenoid valve body and the igniter valve body are part of the main fuel-valve cover. The igniter valve is mechanically linked to the main fuel-valve by a stem and



MAIN FUEL VALVE

Fig. 65

poppet arrangement. A metal bellows is used to ensure a postiive seal between the oxidizer igniter-valve and the main fuel-valve.

The main fuel valve is a normally closed valve due to the spring load applied by the preformed metal disc. The fuel valve opens when fuel-pump discharge-pressure acts upon the metal disc and closes when fuel pump discharge pressure decays.

Figure 66 contains a preliminary-design layout of a novel concept for a main propellant valve. This concept consists of integrating the solenoid control-valve with a main propellant valve. The propellant valve is a unique adaptation of a poppet valve and a butterfly valve.

The butterfly gate rotates to an open position when the piston sleeve moves away from the seat and rotates to a closed position when the sleeve moves towards the seat. The combined translational and rotational motion of the gate is accomplished by integrating a cam follower in the gate pivot and engaging the cam follower in a helical cam groove.

The solenoid valve body is part of the main propellant valve housing. This packaging technique eliminates long feed lines, metal to metal connect points and provides a light weight component assembly. The solenoid valve is of conventional design and is used to pressurize or vent either side of the piston sleeve.

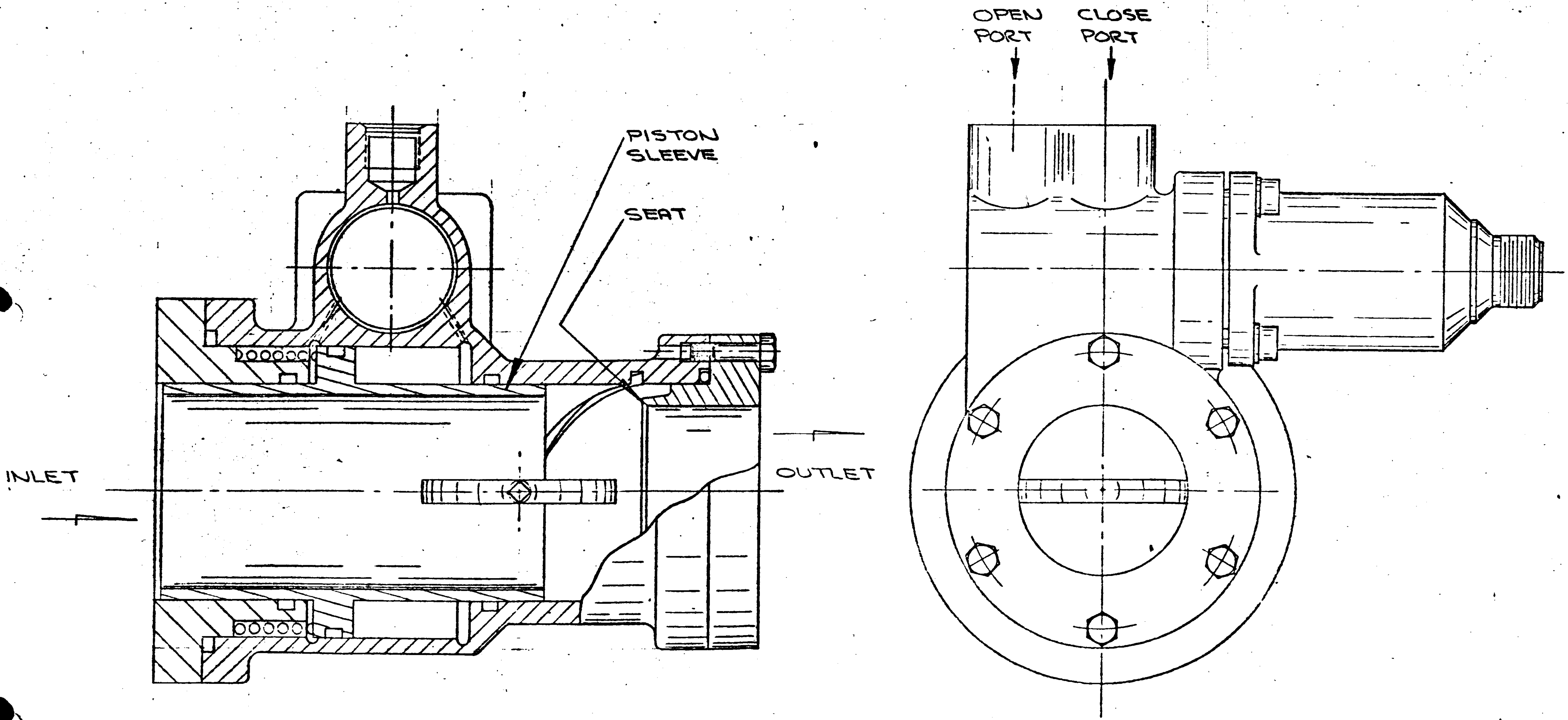
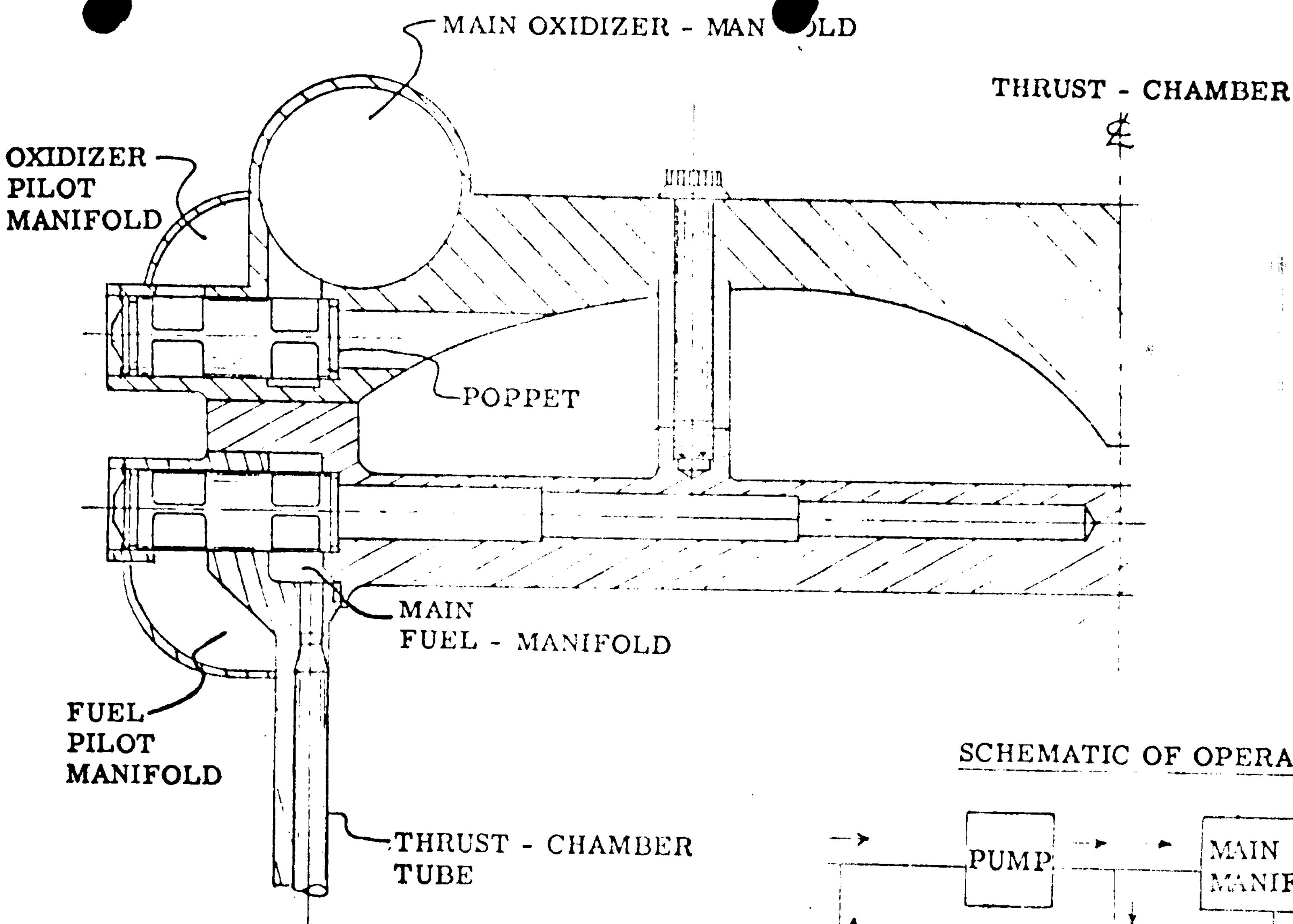


Fig. 66. Main Propellant-Valve Concept

Multiple-Poppet Valve Concept

This design utilizes a number of small cartridge poppet-valves located along the injector manifolds (Figs. 67 through 69). This concept is especially advantageous for large-engine systems having propellant flowrates large enough that use of a single valve for each propellant could present serious packaging and propellant-distribution problems. Fig. 67 contains a schematic which indicates how these valves are operated within a system. Propellant at [illegible] passes through the normally-open ports of the pilot valve and to the pilot manifold, where it holds (at [illegible] in Fig. [illegible]) the poppets in the closed position, since the pressure in the main manifold is acting on a smaller area (C of Fig. 69) of the poppet. When the pilot valve is actuated the closing pressure in the pilot manifold is vented to the pump inlet, and pressure in the main manifold is now able to move the poppets outward, thus opening the valves. In addition to its compactness, this concept has the advantage that no dynamic seals are required, since any leakage past the poppets is simply returned to the pump inlet.

Operation of this concept has been described in terms of a single propellant; it should be noted that the mode of operation is identical for the other propellant of the system.



RADIAL CARTRIDGE POPPET-VALVES

SCHEMATIC OF OPERATION

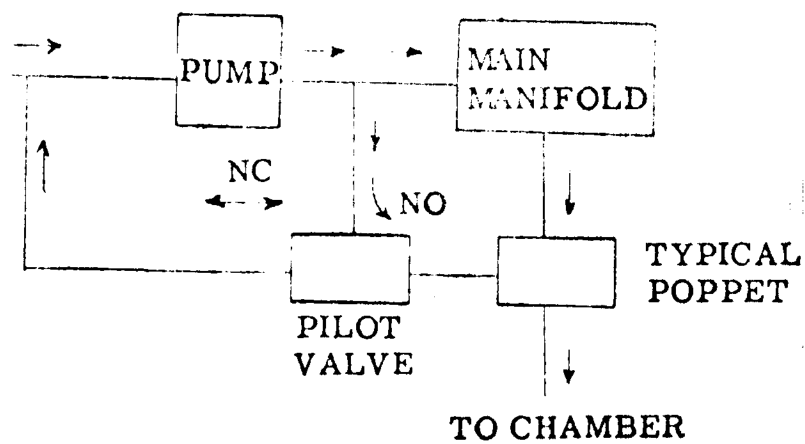


Fig. 67

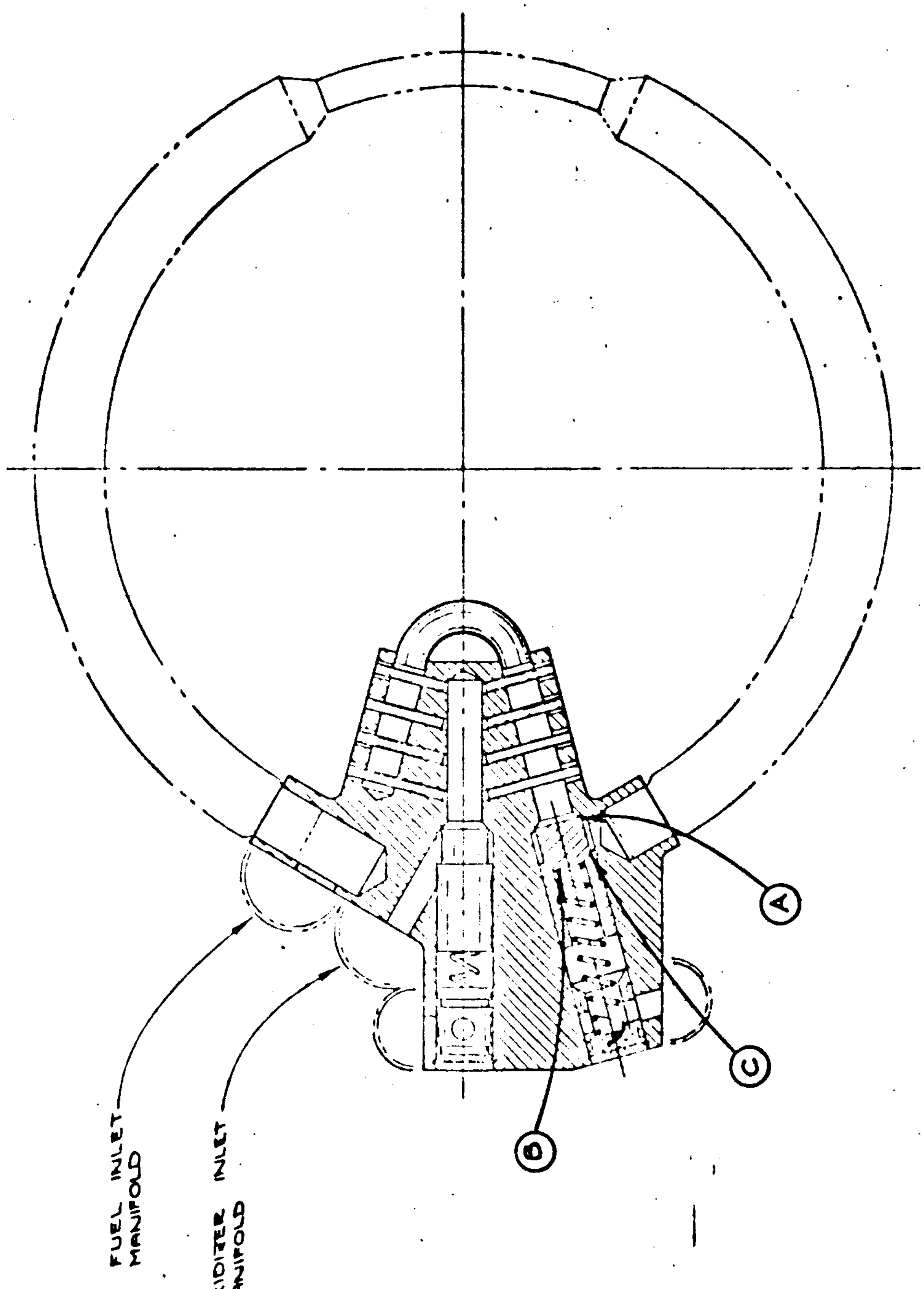


Fig. 68. Poppet-Valve Installation

Figure 68 shows how this concept could be used with a toroidal combustion chamber; this is also shown as an incidental item in Figures 73 and 74.

Figure 69 depicts a typical installation of this concept on a conventional thrust chamber.

This concept is applicable to all the propellant combinations considered as a part of this program. As implied previously, it is probably beneficial only for larger-thrust engines.

Propellant-Valve/Turbopump Concepts. These concepts are similar to the multiple poppet-valve concept in that they too are better suited to larger-thrust systems. Two concepts are shown in Figures 70 and 71. The characteristic feature of both of these concepts is multiple flow-passages, which are distributed along the periphery of the pumps, thus eliminating the large propellant valves which are characteristic of conventional systems wherein only one flow passage from each pump is provided.

The first of these concepts, termed the circular-gang quarter-turn ball valve (Fig. 70), essentially consists of a number of ball valves which are linked together. Valve-actuation is effected by rotation of a single "control" ball.

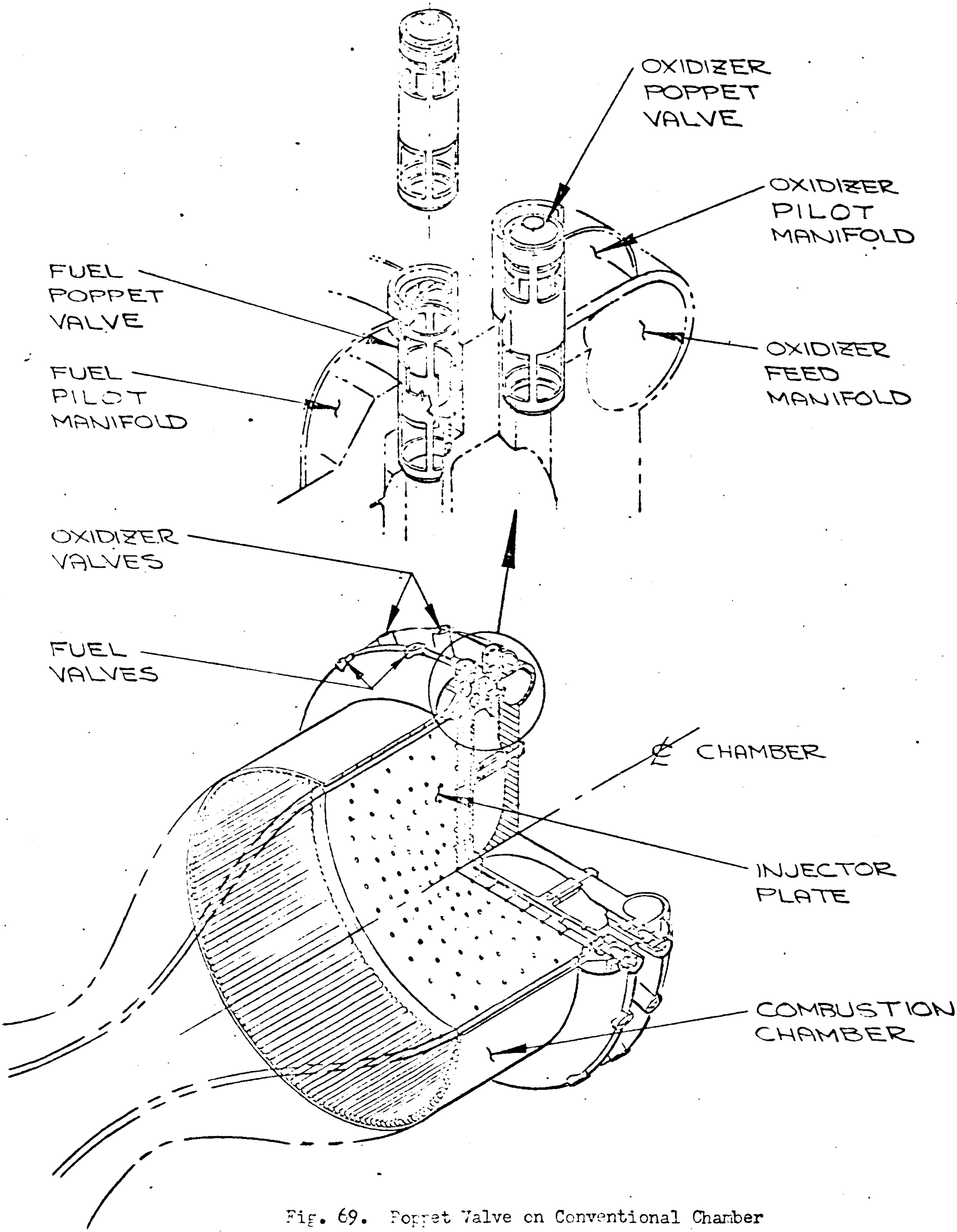


Fig. 69. Poppet Valve on Conventional Chamber

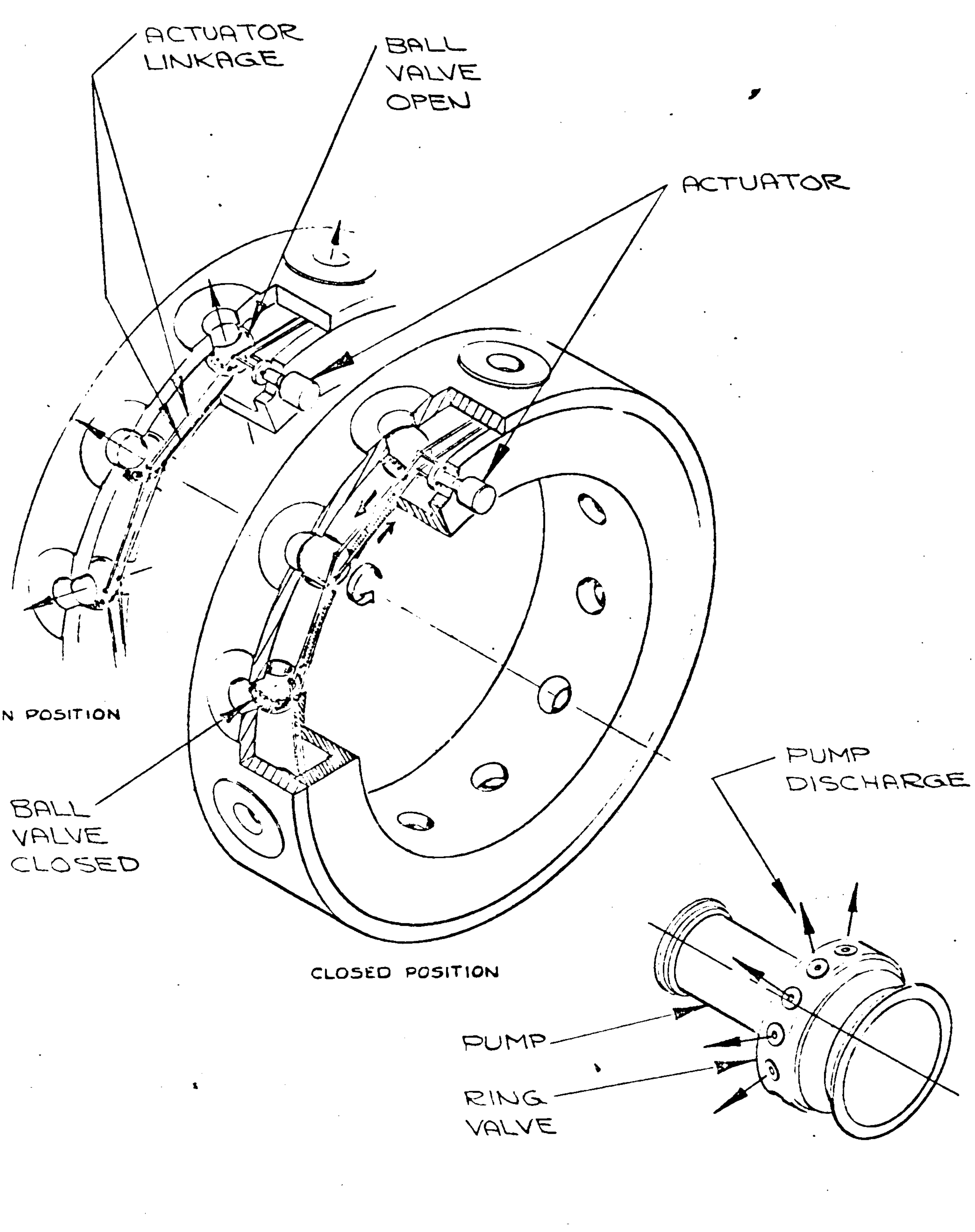


Fig. 70. Circular-Gang Ball-Valve

The second concept, the turbopump ball-bearing valve (Fig. 71), consists of a number of loosely retained balls. Valve closure is effected by rotating the ball retainer, which depresses the undulate spring slightly, allowing the ball to roll into the outlet port and shutting off the flow. Opening of the valve is performed similarly.

Each of these Figures (70 and 71) contains an illustration showing how this type of valve could be installed on a turbopump. The use of this concept with the "cartridge" turbopump concept is shown in the next section.

Cartridge Integration Concept. The cartridge concept essentially consists of designing components so they can be "plugged" into a hole. These holes are parts of a common manifold and structural member which has built-in passages for the propellant flow. Thus, this concept tends to reduce system size and weight by having components "share" structural elements.

Figure 72 shows the use of this concept in a multiple turbopump system with a bell thrust-chamber.

Figure 73 depicts the use of a cartridge turbopump with an advanced annular combustion chamber. The adaptability of this concept to modular packaging of a multiple-turbopump system is shown in Figure 74 .

This concept is not limited in application with regard to propellants or thrust level.

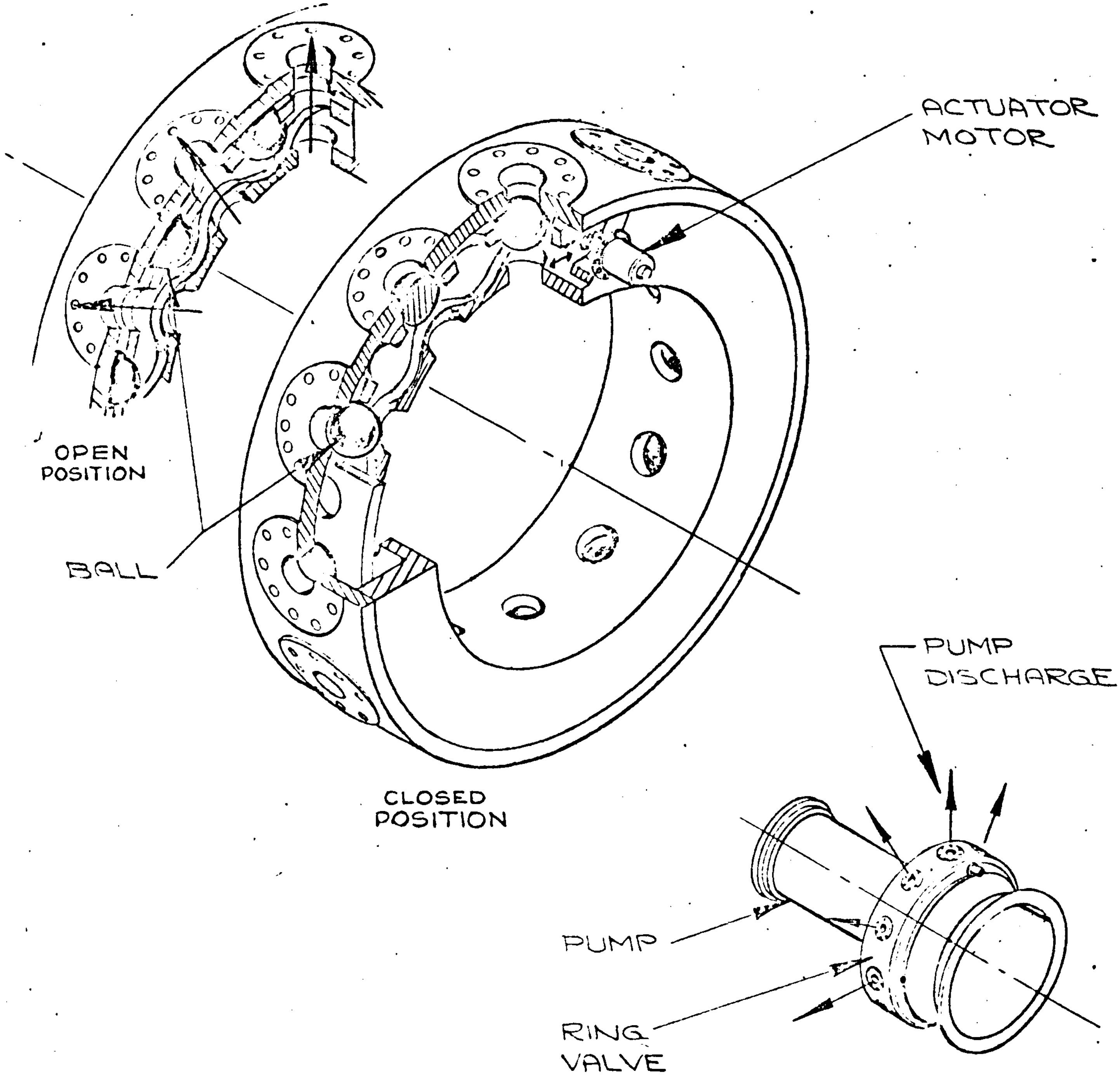


Fig. 71. Ball Bearing Valve Concept

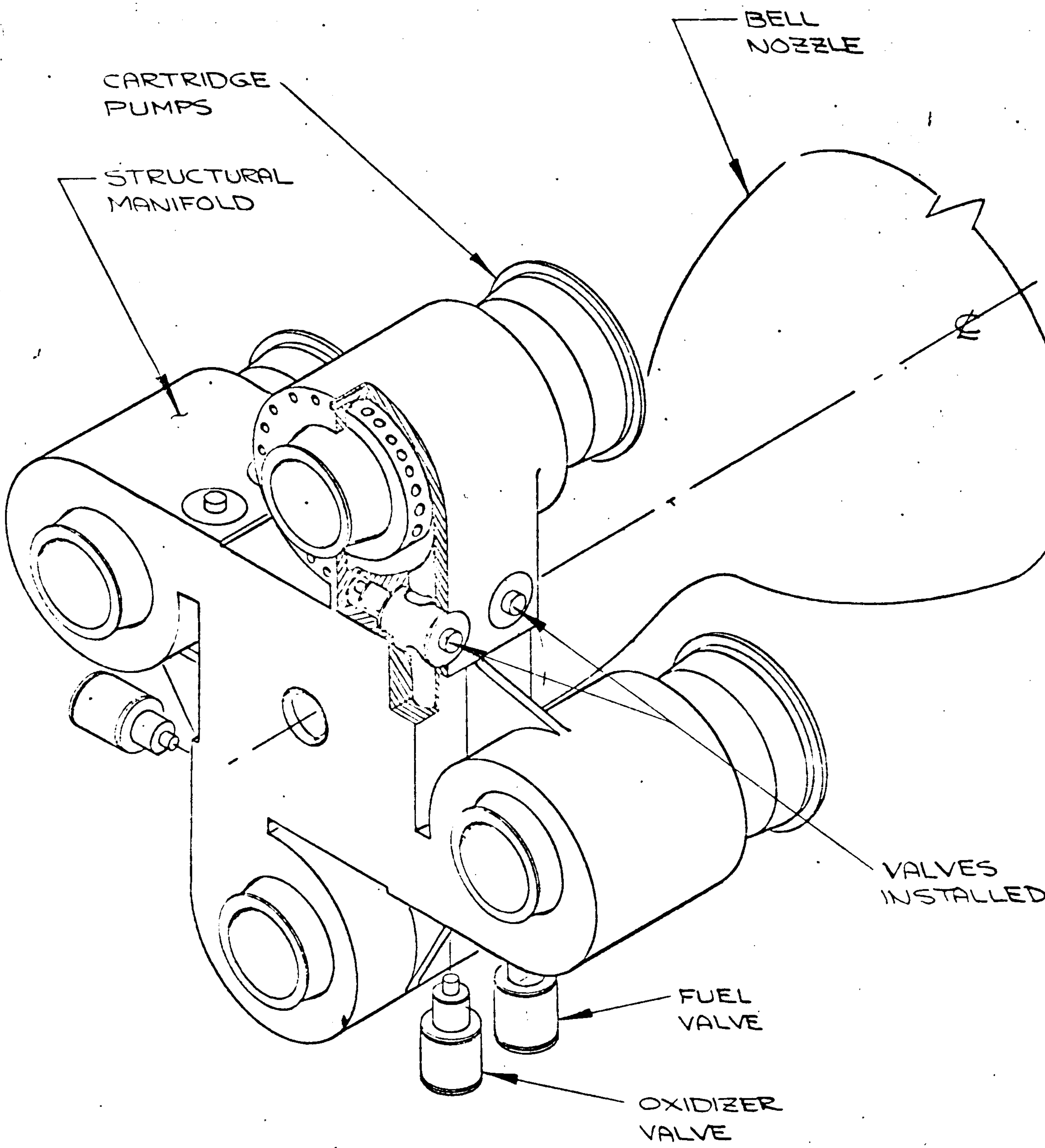
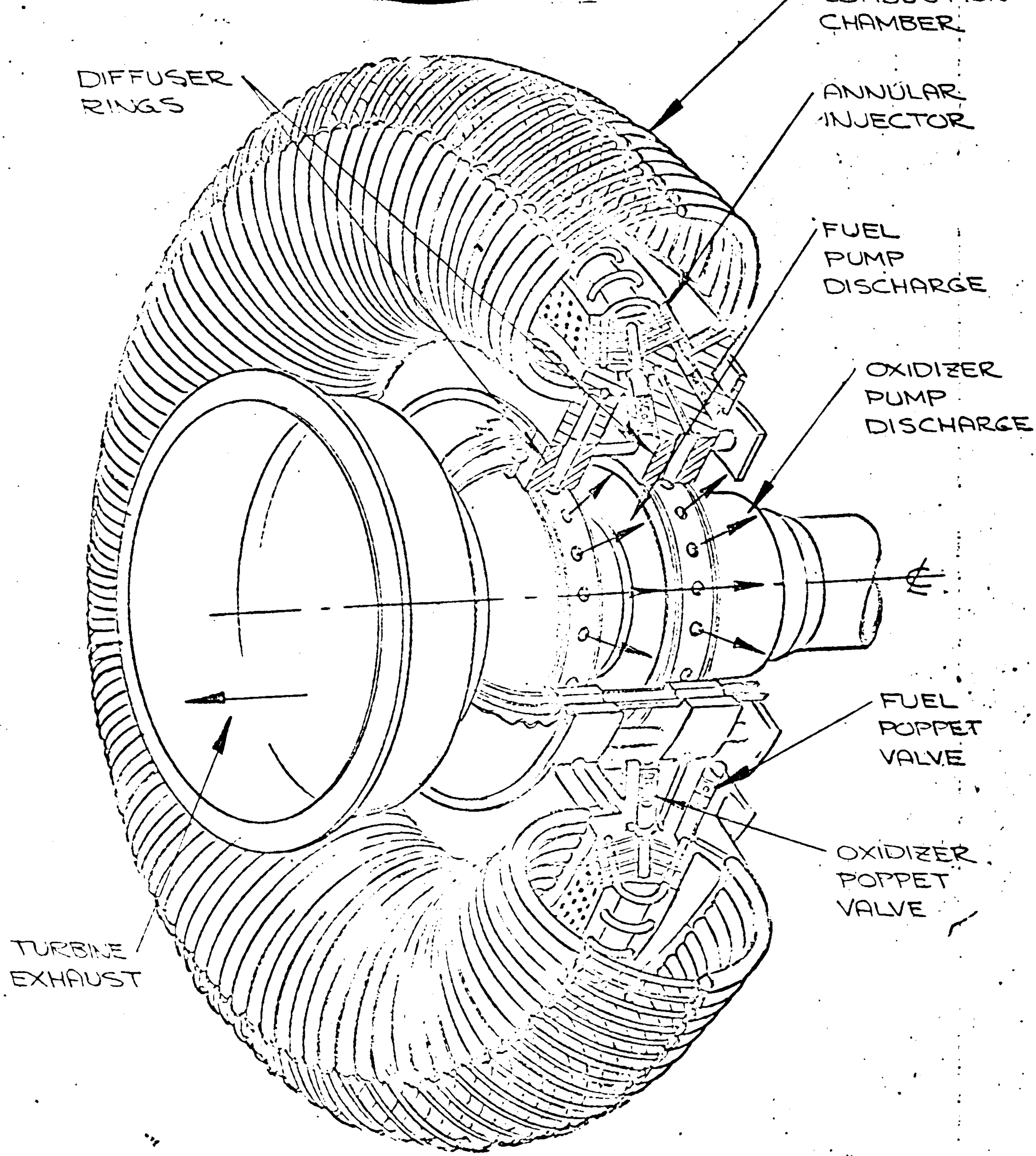


Fig. 72. Cartridge Turbopumps with Conventional Nozzle

[redacted]



[redacted]

Fig. 73. Cartridge Turbopump with Annular Combustor

[redacted]

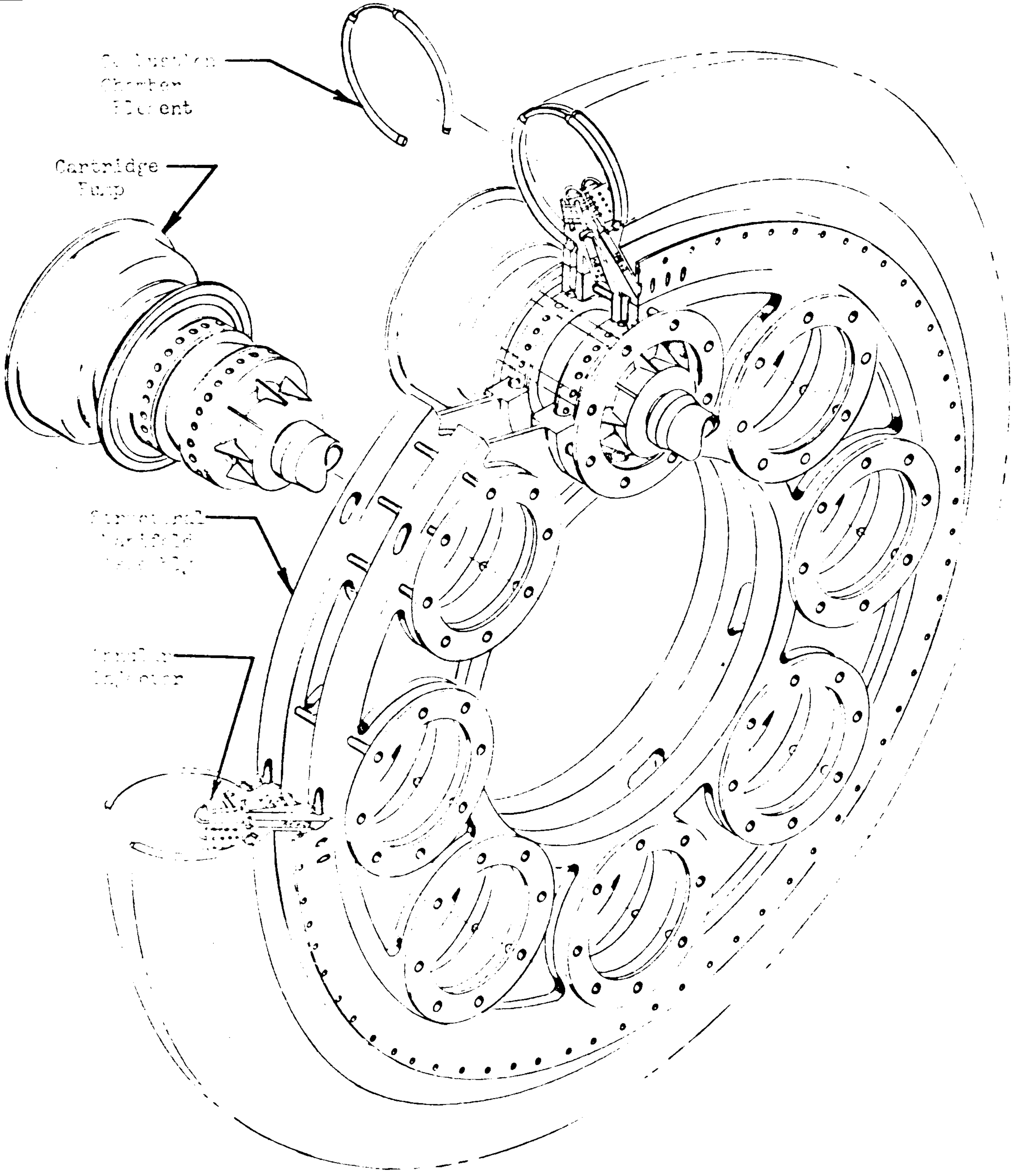


Fig. 74. Multiple-Pump Cartridge-Turbopump Concept

Start System Integration. Two highly-integrated start-system concepts have been evolved for use in a multiple-start NTO/50-50 spacecraft engine (system 302, Fig. 14). These systems have been sized on the basis of five starts. Both start-systems utilize a bi-propellant gas-generator.

The unitized gas-generator start-system (Fig. 74) integrates most of the engine start-system into one integrated assembly. The main valves and the tapoff check valves are combined with the injector. The oxidizer start-tank is spherical and is mounted on the injector as shown. The fuel start-tank is an annular tank which surrounds the body of the gas-generator.

Propellants are delivered to the injector using vehicle-supplied high-pressure helium. Positive-displacement diaphragms isolate the propellants from the pressurant. A porous shell in the bottom half of each tank prevents trapping of propellants.

One control port is required for operation. When gas enters this port the propellants become pressurized and the main valves open.

The integrated turbine-spin start-system (Fig. 75) is a compact, highly integrated start-system which could utilize the turbine inlet manifold as the gas-generator combustor (Fig. 76). The engine start-tanks and the main valves are integrated into one assembly which bolts onto the turbine

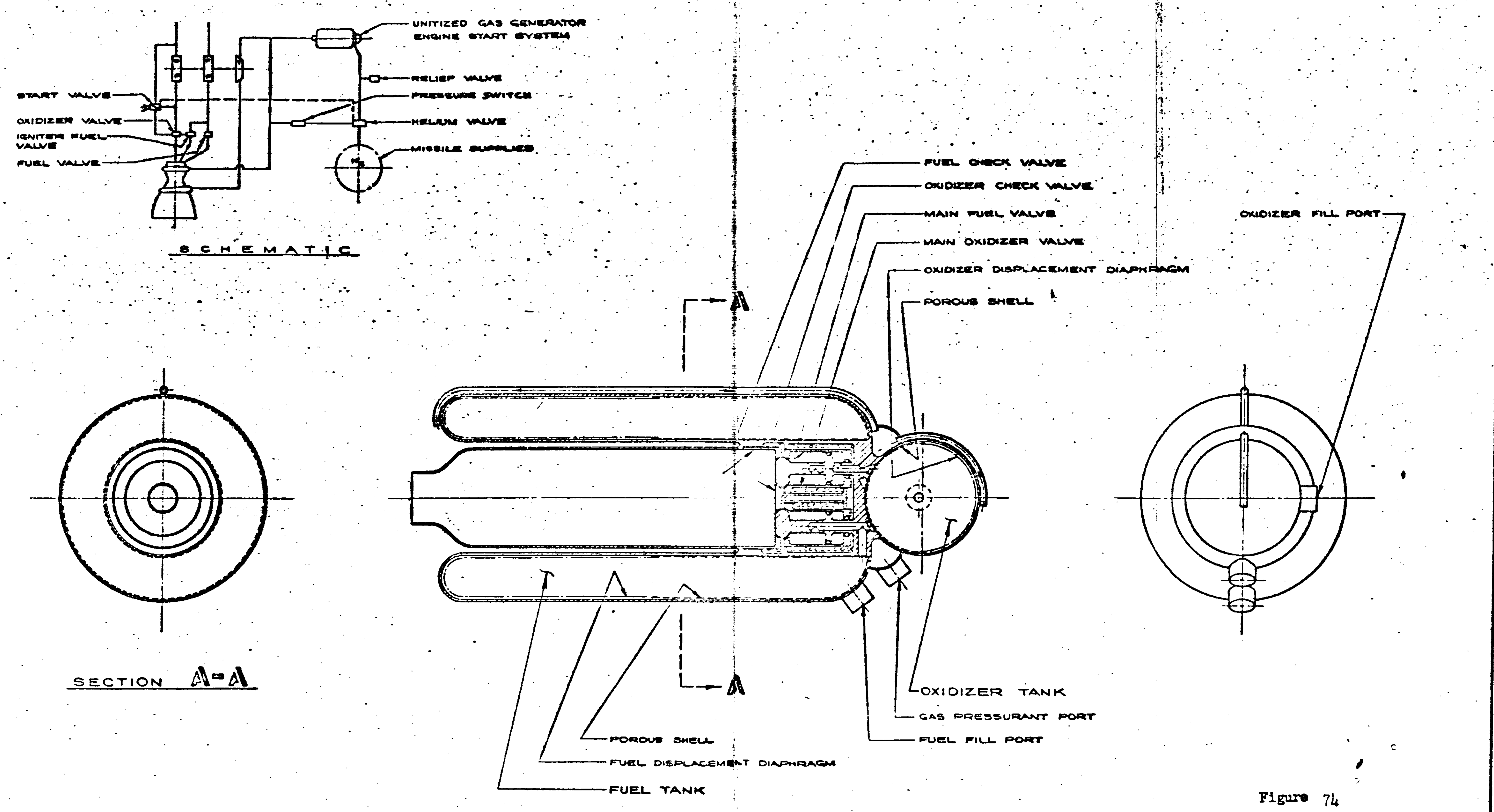


Figure 74

ADVANCED DESIGN

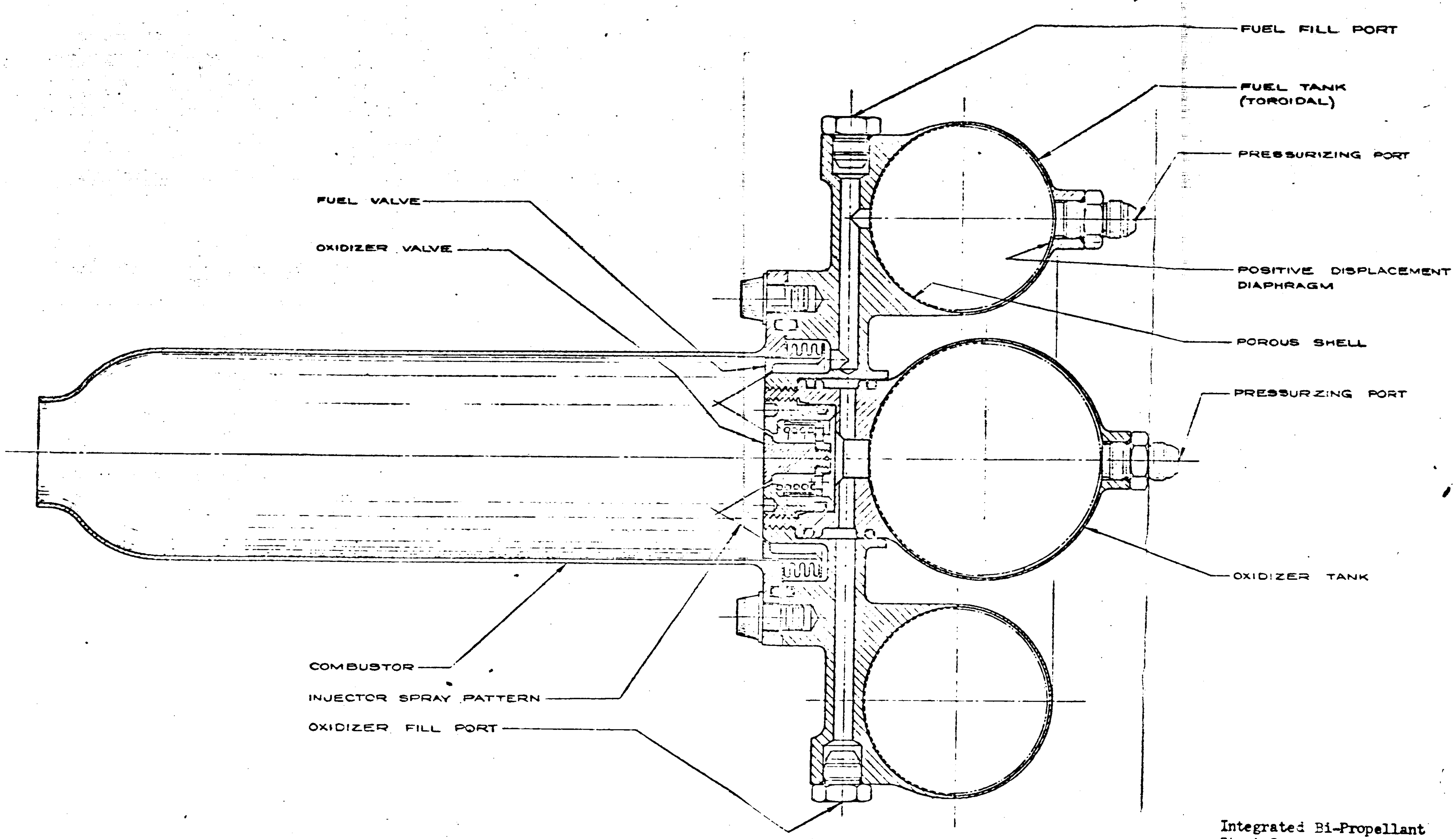
ROCKETDYNE
A DIVISION OF NORTH AMERICAN AVIATION, INC.

UNITIZED GAS GENERATOR
ENGINE START SYSTEM

T.E. COWELL

FULL

2277



Integrated Bi-Propellant Start System

Figure 75

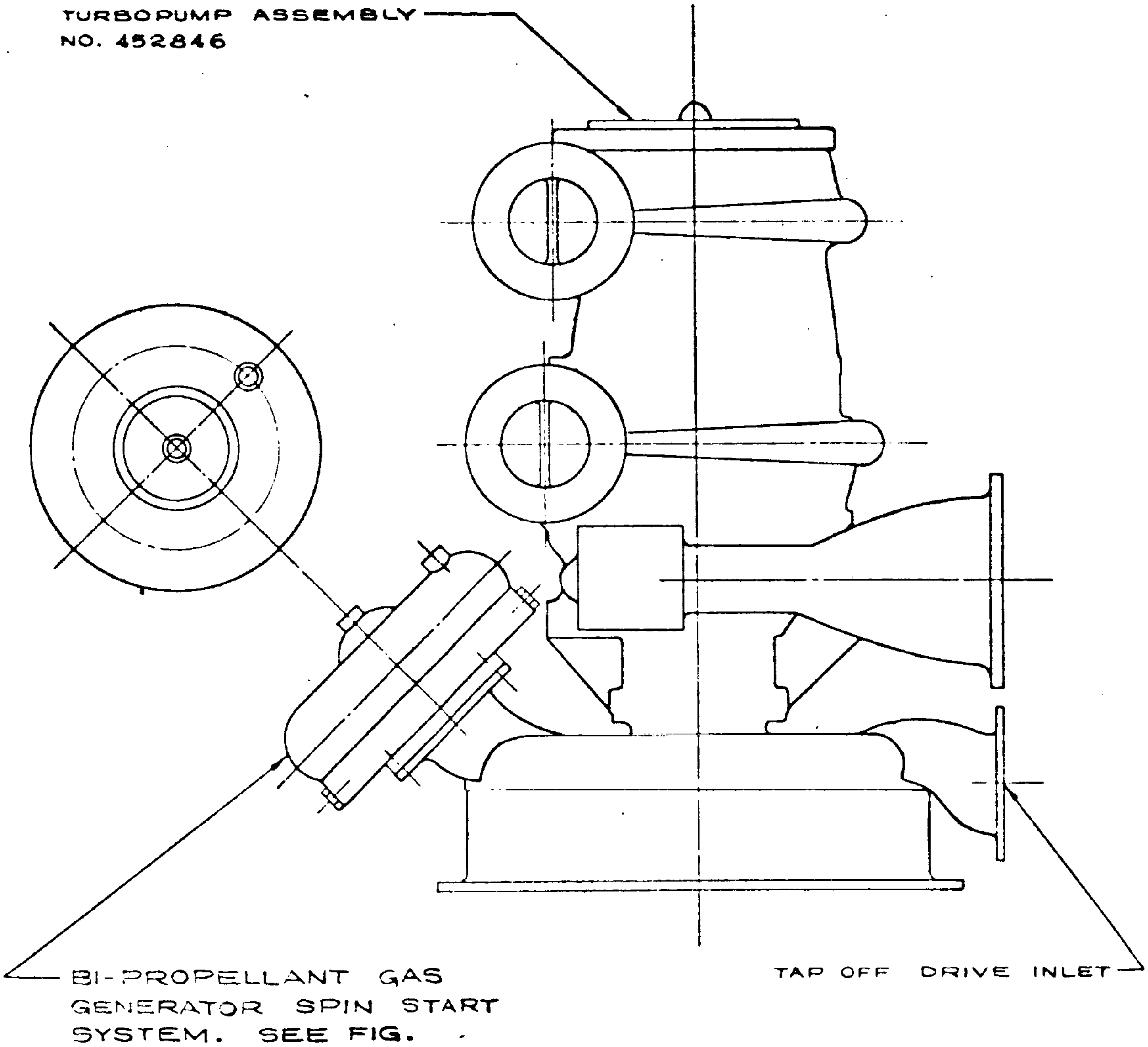


Figure 76

ADVANCED DESIGN

ROCKETDYNE
A DIVISION OF NORTH AMERICAN AVIATION INC

TURBINE SPIN START
INSTALLATION

T.E.COWELL

HALF SIZE | APR. 17, 1964

2348

inlet manifold. The main valves are relief-check type valves. When pressure is applied to the start tanks, the main valves open and ignition is accomplished, since the propellants are hypergolic.

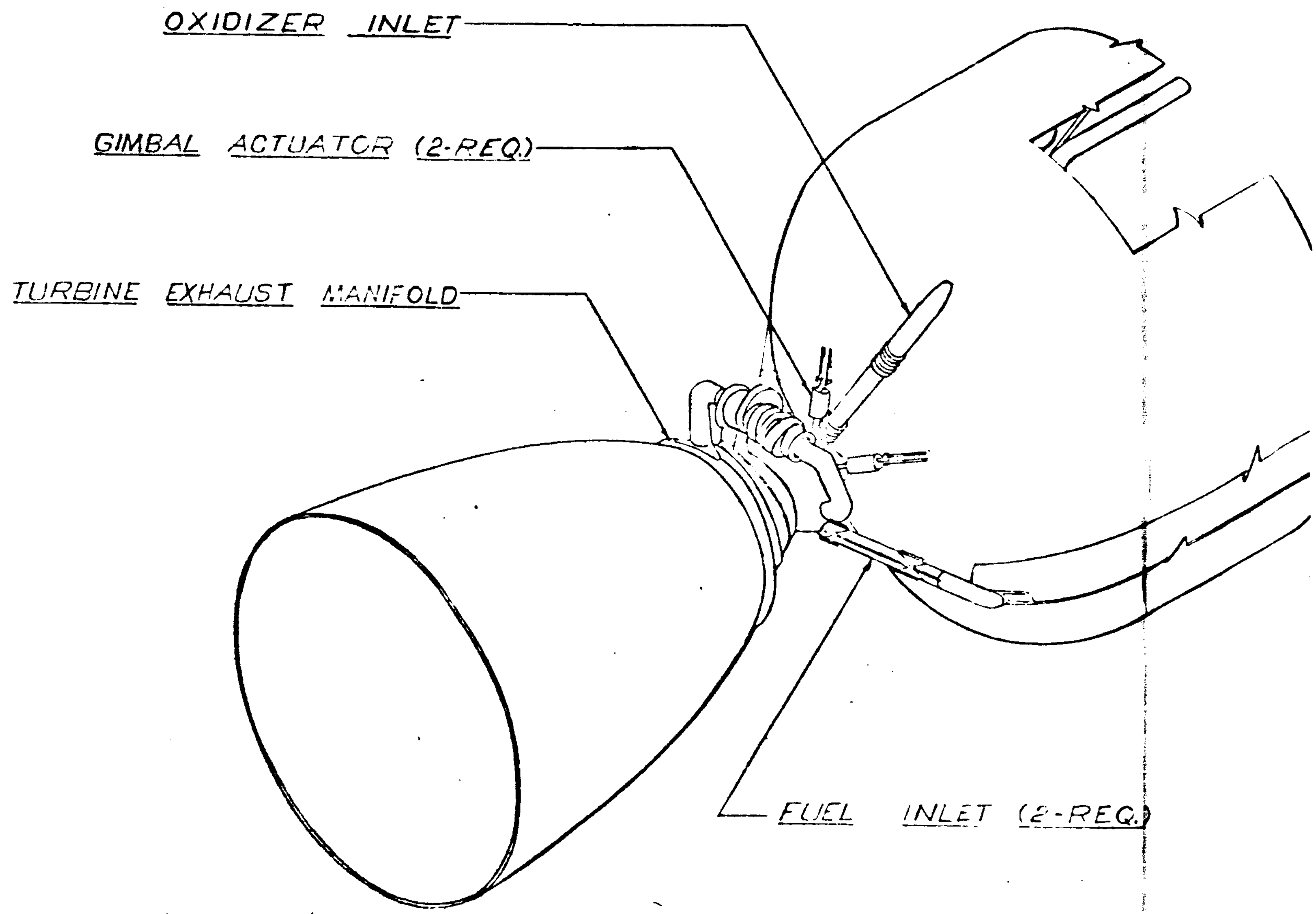
When tap-off pressure builds up in the tap-off line, the main valves close, since they are check valves.

This system could be designed with rechargeable propellant tanks which would make it smaller while providing unlimited restart capability. This would, however, involve a reliability degradation (see the section on system ratings).

These concepts are, of course, limited to hypergolic propellant combinations, but could with slight modification, be adapted to non-hypergolic combinations.

Three-Leg Gimbal. The three-leg gimbal (Fig. 77) concept integrates the gimbal bearing, thrust structure, and propellant inlet ducts. This places the gimbal point closer to the engine center of gravity, thus reducing the required actuator loads.

The over-all features of this concept are described, with regard to Figure 50, in the section on spacecraft using conventional nozzles. In this section, some of the details of the concept are discussed, namely, (1) canted bellows, (2) ball joints for thrust transmission, and (3) pinned



Three-Leg Gimbal Concept

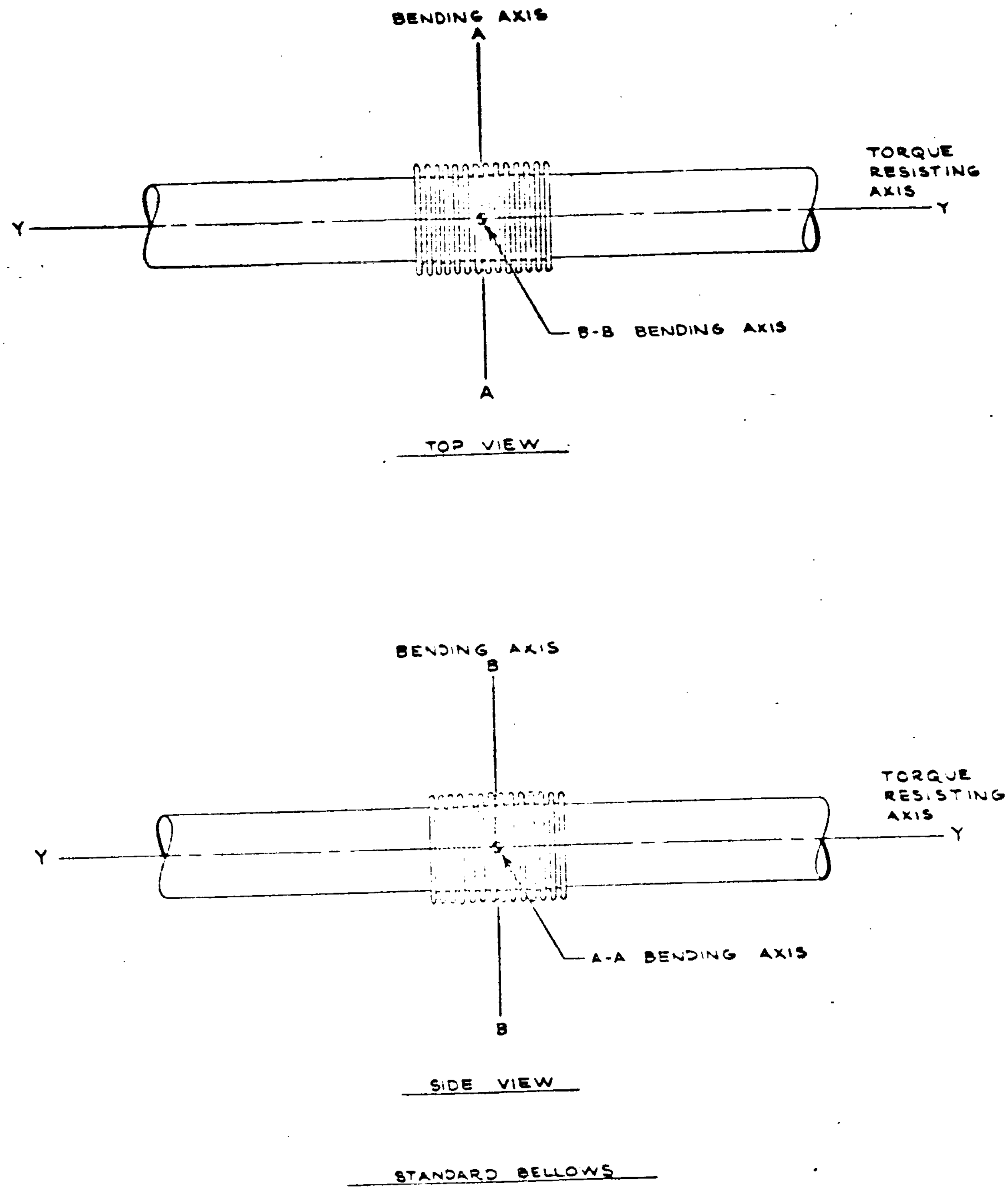
Figure 77

joints for thrust transmission.

The purpose of the canted bellows is to allow a flexible joint in a propellant line to bend about axes other than the ones perpendicular to the line axis. Figure 78 shows a conventional bellows and line arrangement. The propellant line and torque resisting axis is Y-Y. The propellant line can bend about axis A-A and B-B without rotating one half of the line relative to the other. The bellows acts as a constant-velocity universal-joint. Assume it is desired to keep the same bending axes A-A and B-B and torque resisting axis Y-Y, but change the propellant lines axis to X-X as shown in Figure 79 . This configuration allows the propellant line to be rotated relative to the torque resisting axis Y-Y. The bending and torque-resisting axes can now be set up independent of the direction of the propellant line, thus allowing greater packaging freedom.

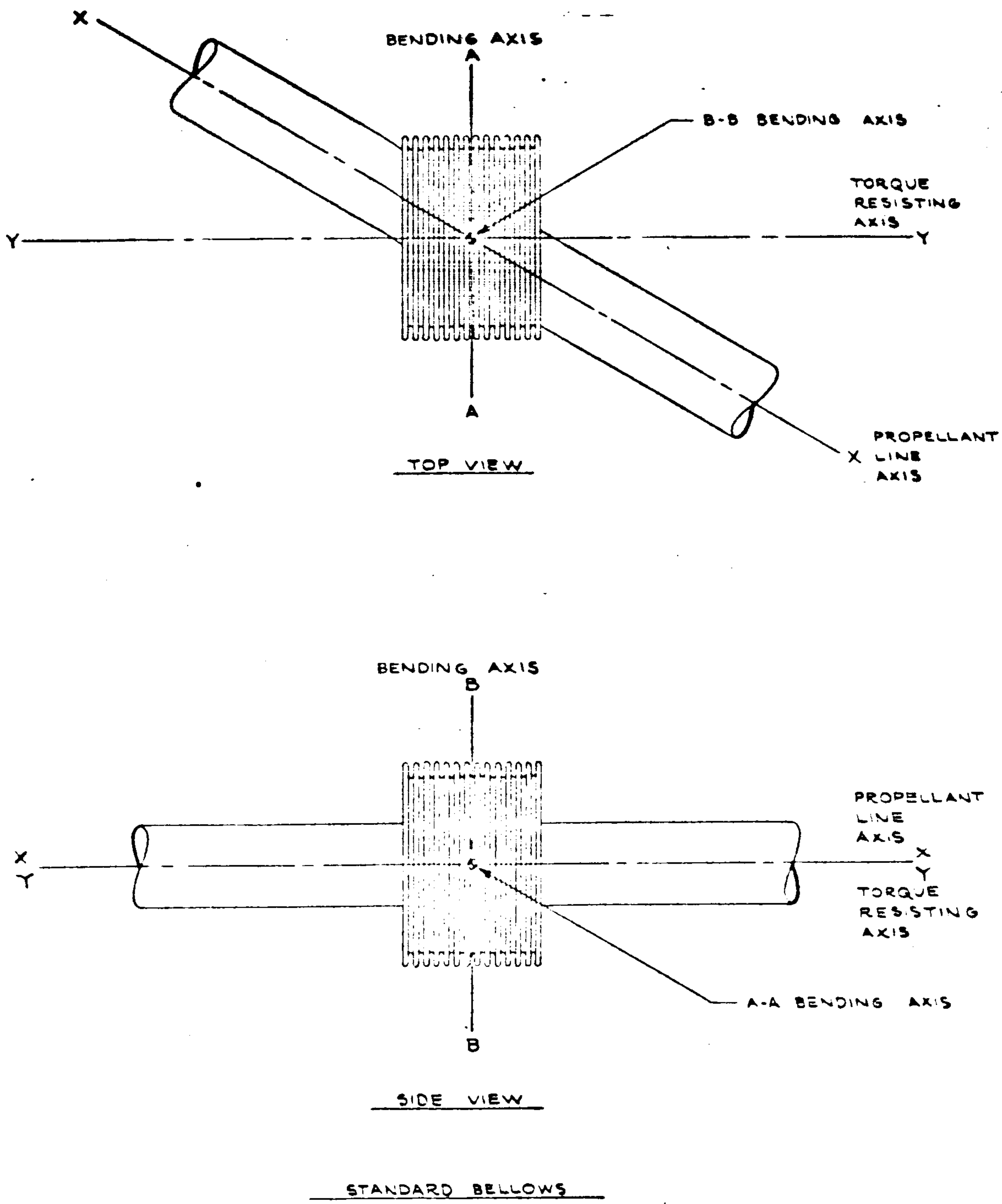
The bellows shown in Figure 79 are comparatively large. To reduce the bellows size, the canted bellows shown in Figure 80 are used. These bellows combine the smooth propellant flow and small size of the Figure 78 bellows and the packaging freedom shown in Figure 79.

A brief discussion of how the canted bellows can be used to simplify the inlet-duct configuration for systems using other gimbaling methods, appears in the section on spacecraft systems with advanced nozzles.



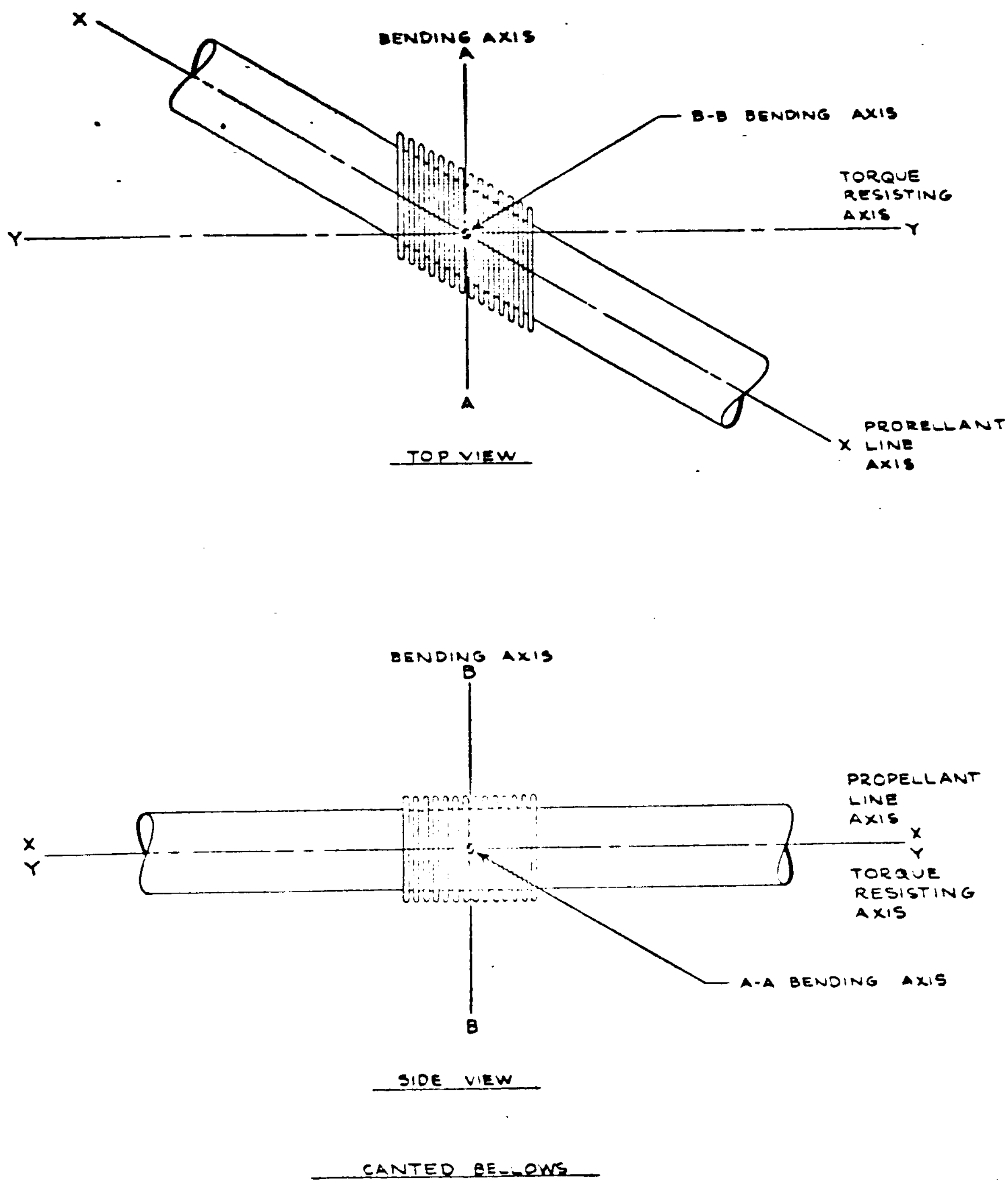
Standard Bellows, Conventional Installation

Figure 78



Standard Bellows, Canted Installation

Figure 79



Canted Bellows, Installation

Figure 80

Figure 81 contains a layout of one of the three ball joints that could be used to transmit the thrust while allowing the engine to gimbal. The ball joint shown in Figure 82 could be used instead, if it were desirable to isolate the two moving-contact surfaces from the propellants.

Figure 83 depicts an alternate approach, the pinned joint, or cardan ring. This concept removes the bearing surfaces, and provides better access for gimbal-bearing servicing.

As mentioned previously, this concept has no real applicability limitations with regard to propellants or thrust level, but heat-transfer problems associated with the use of multiple inlet ducts may make it less desirable for systems using cryogenics.

Tubular Spherical-Combustor. The tubular spherical-combustor concept (Fig. 84) is similar to the toroidal-combustor concept (Ref. 12). This concept may be especially applicable to low-thrust annular engines.

Figure 85 indicates how this spherical combustor could be used as a thrust-vector-control device with a fixed thrust chamber. The combustor would be gimbaled while the thrust chamber remains fixed.

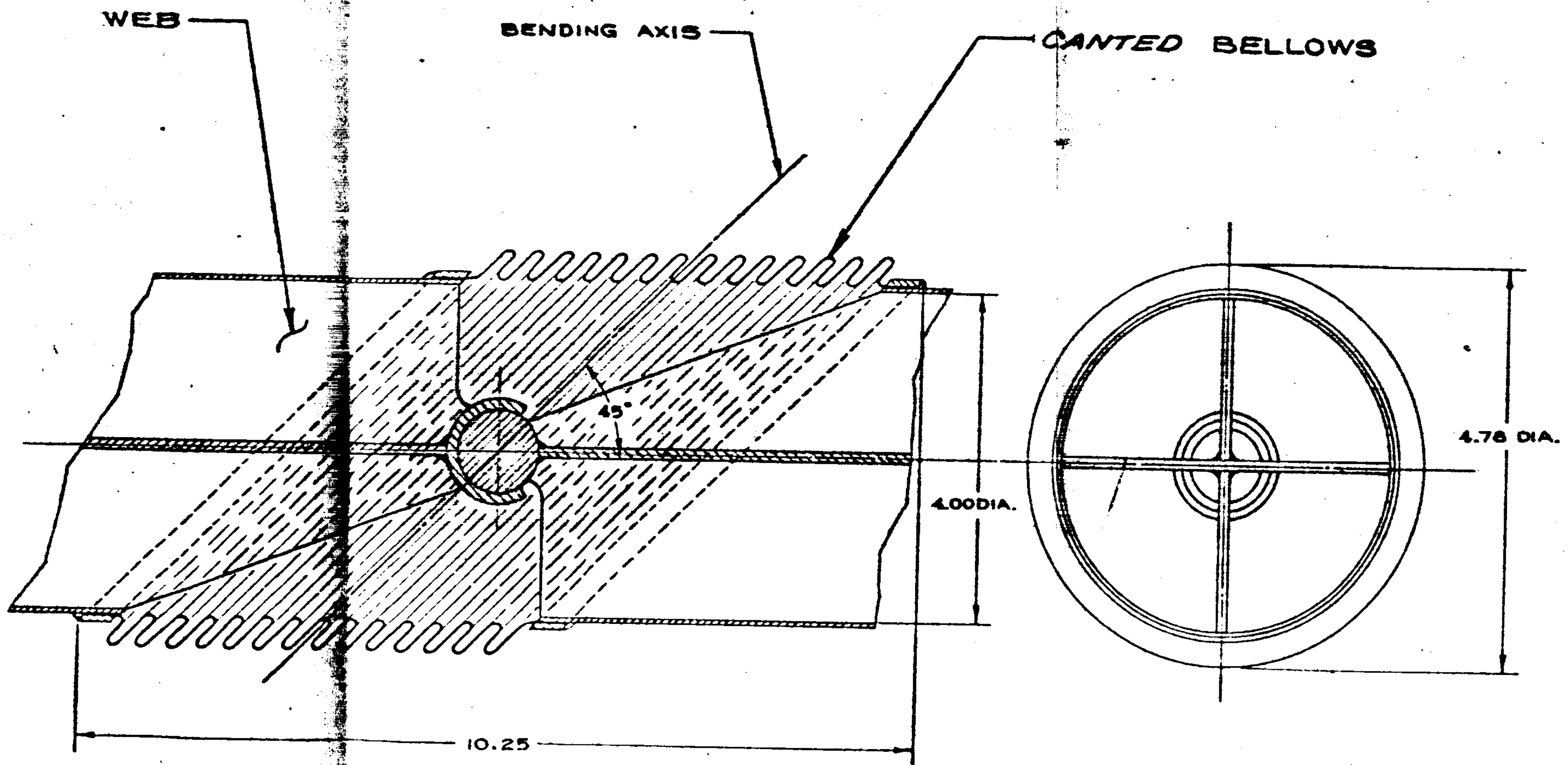


Figure 81

ADVANCED DESIGN

ROCKETDYNE
DIVISION OF NORTH AMERICAN AVIATION, INC.

INLET LINE
FLEX JOINT

T. E. COWELL
SCALE FULL SIZE DATE APRIL 15, 1964

RAP-2345

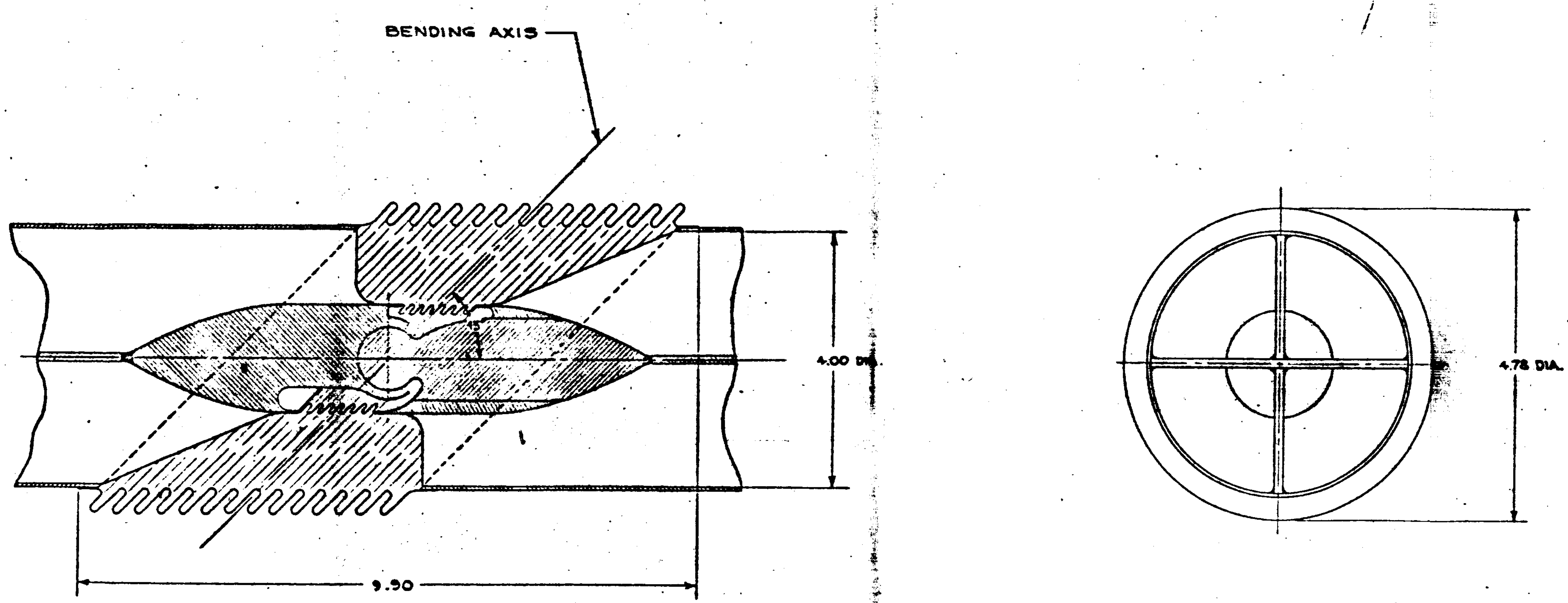


Figure 82

ADVANCED DESIGN

ROCKETDYNE
A DIVISION OF NORTH AMERICAN AVIATION, INC.
6633 CANOGA AVE.
CANOGA PARK, CALIFORNIA

INLET LINE
SEALED BALL JOINT

L H RUSSELL
SCALE: FULL
DATE 5-13-64

AP-2368

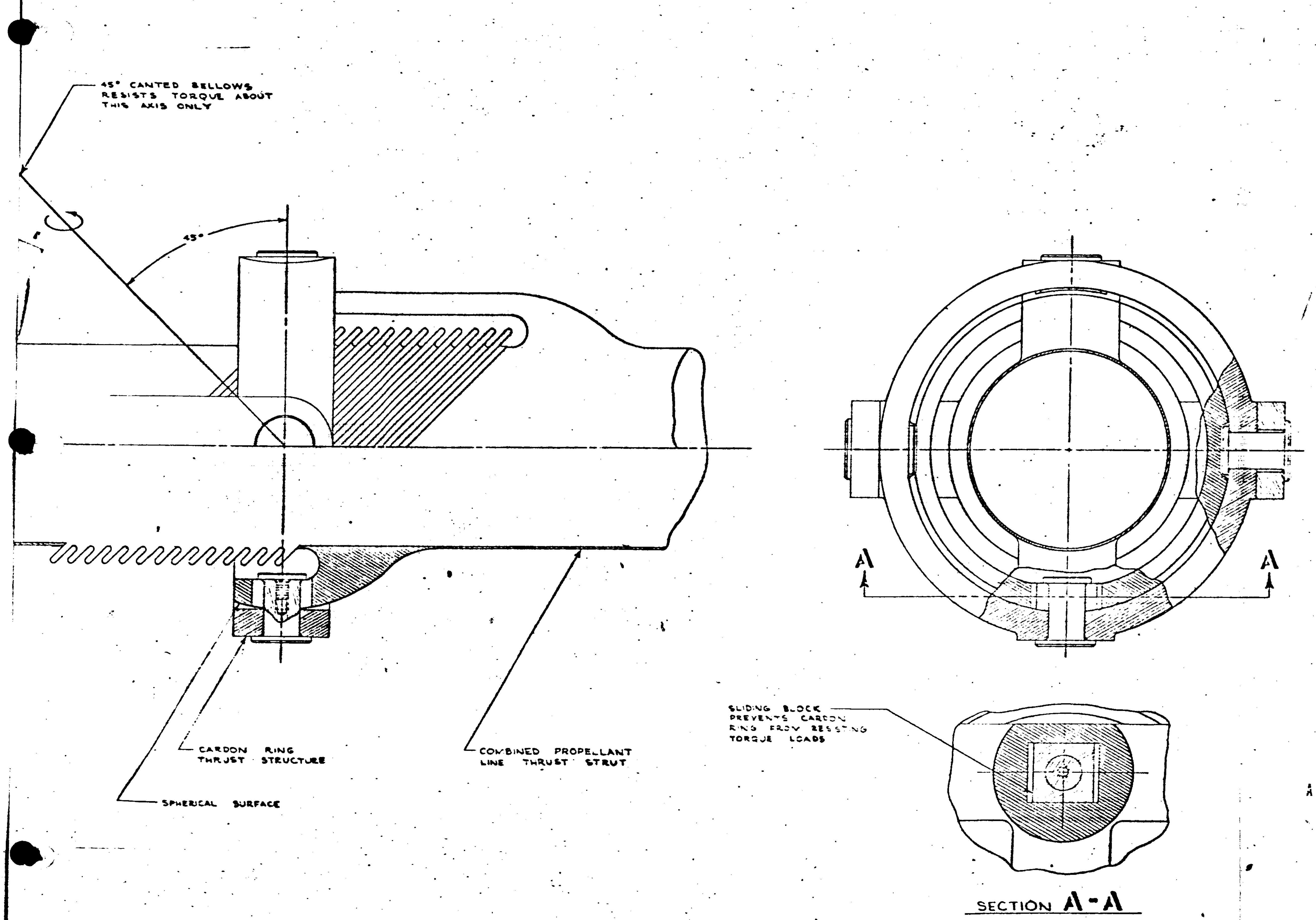
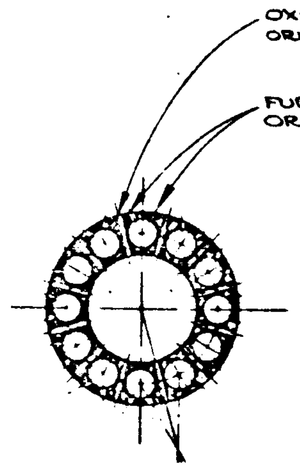


Figure 83

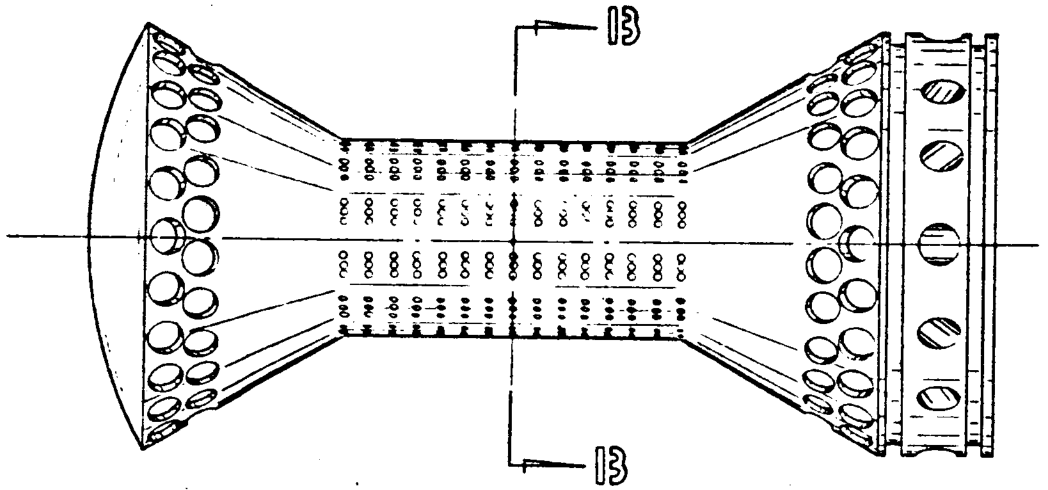
ADVANCED DESIGN

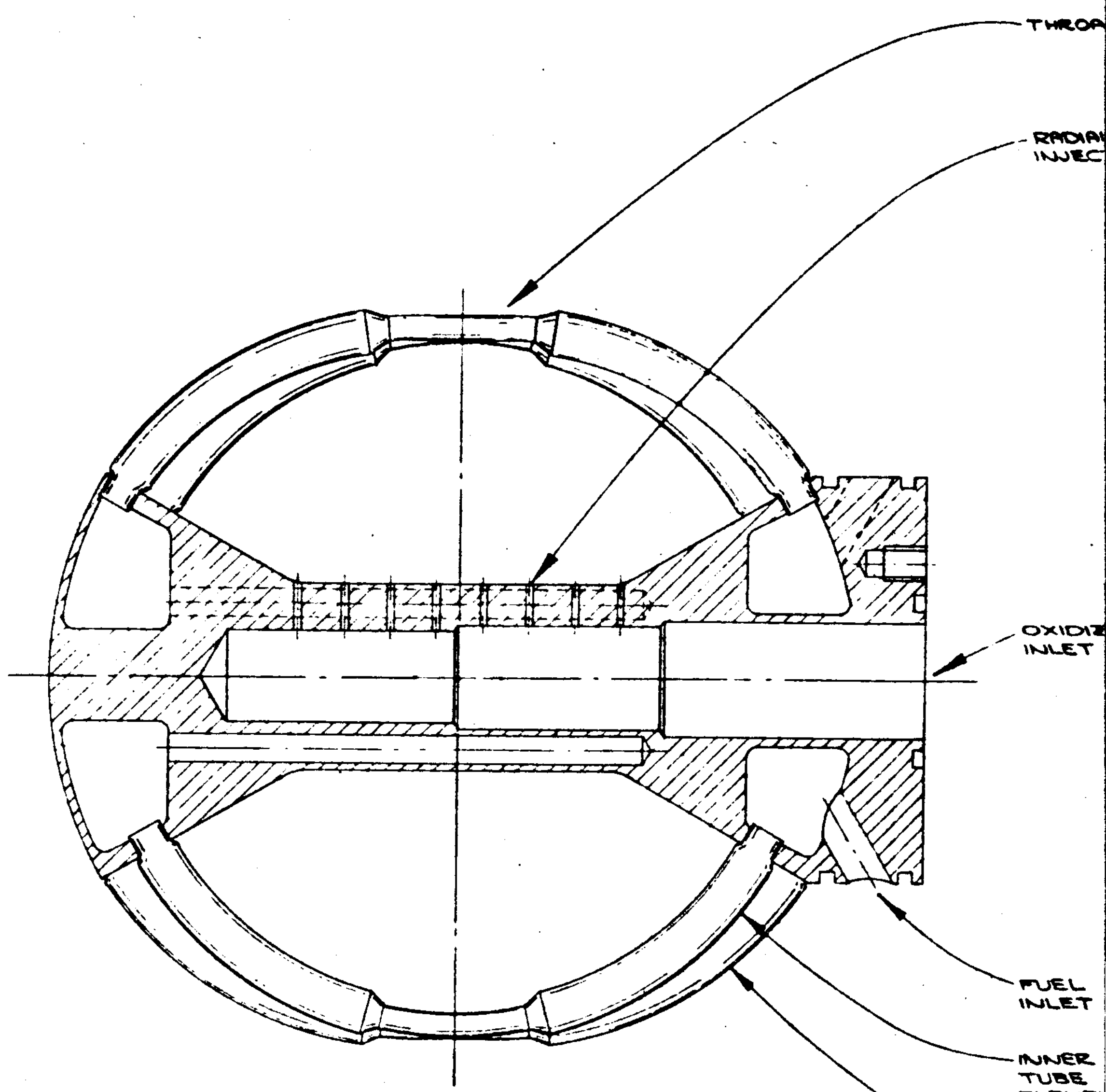
ROCKETDYNE
A DIVISION OF NORTH AMERICAN AVIATION, INC.
CARD N RING - CANTED BELLOWS GIMBAL BEARING
SCALE FULL DATE 4-17-64
AP-2347

[redacted]



SECTION B-B





SECTION A-A

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FOLD-OUT #1

[redacted]

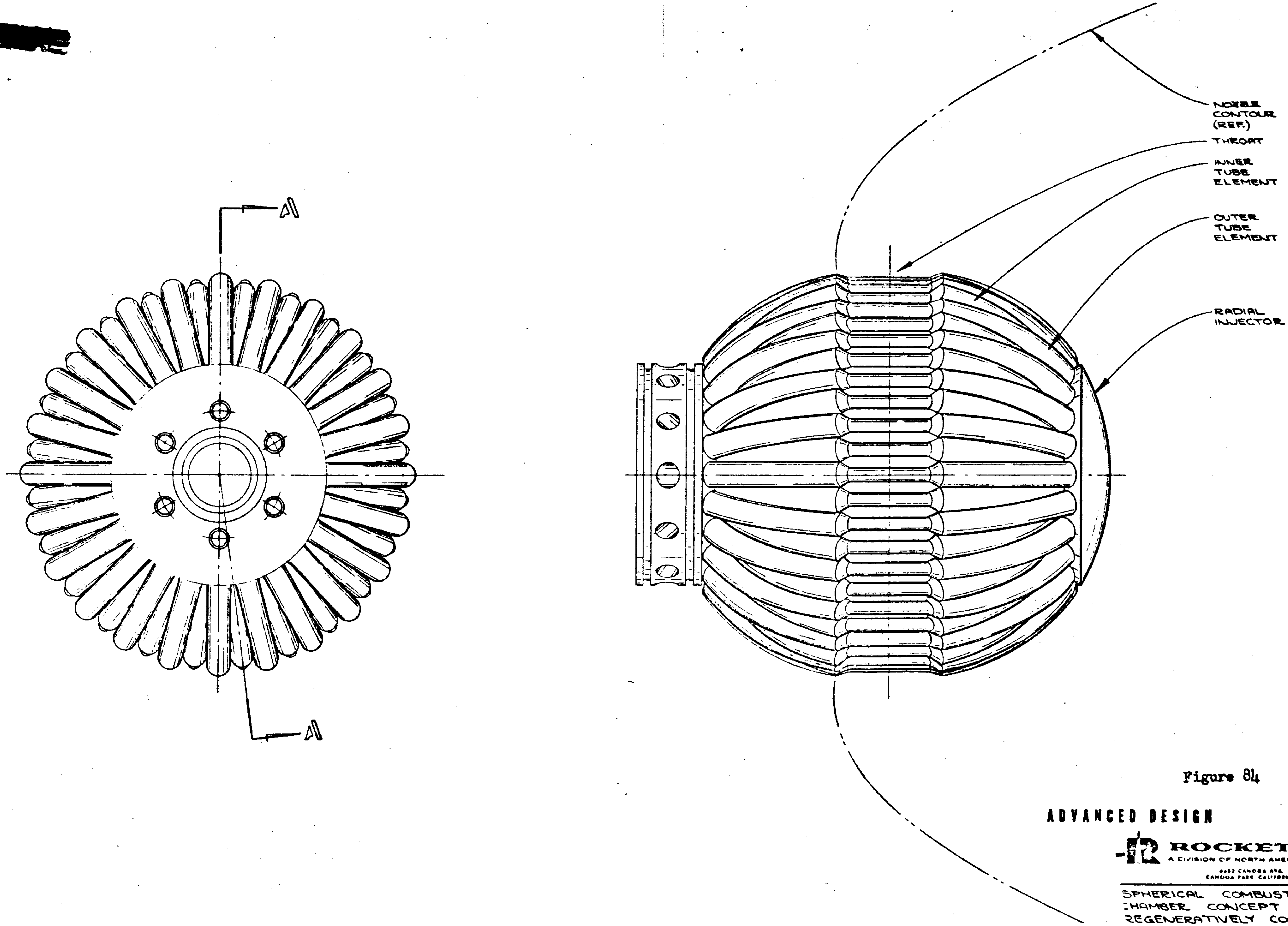


Figure 84

ADVANCED DESIGN

ROCKETDYNE
A DIVISION OF NORTH AMERICAN AVIATION, INC.
6633 CANOGA AVE. CANOGA PARK, CALIFORNIA

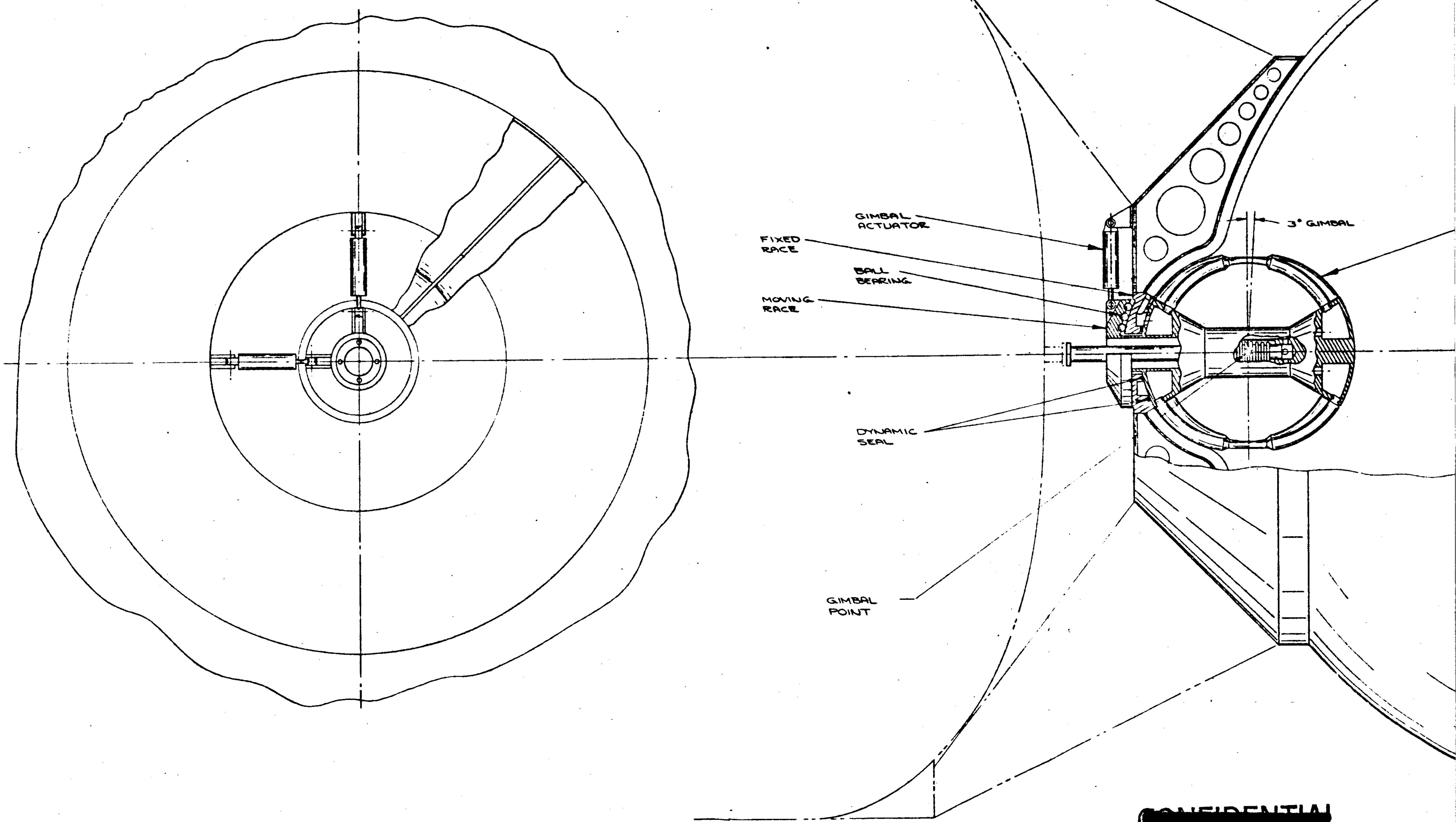
SPHERICAL COMBUSTION
HAMBER CONCEPT :- TUBULAR
REGENERATIVELY COOLED

OCHE JR.
NONE | DATE 3-20-64

AP-2326

FOLD-OUT #2

[redacted]



~~CONFIDENTIAL~~

FOLD-OUT #1

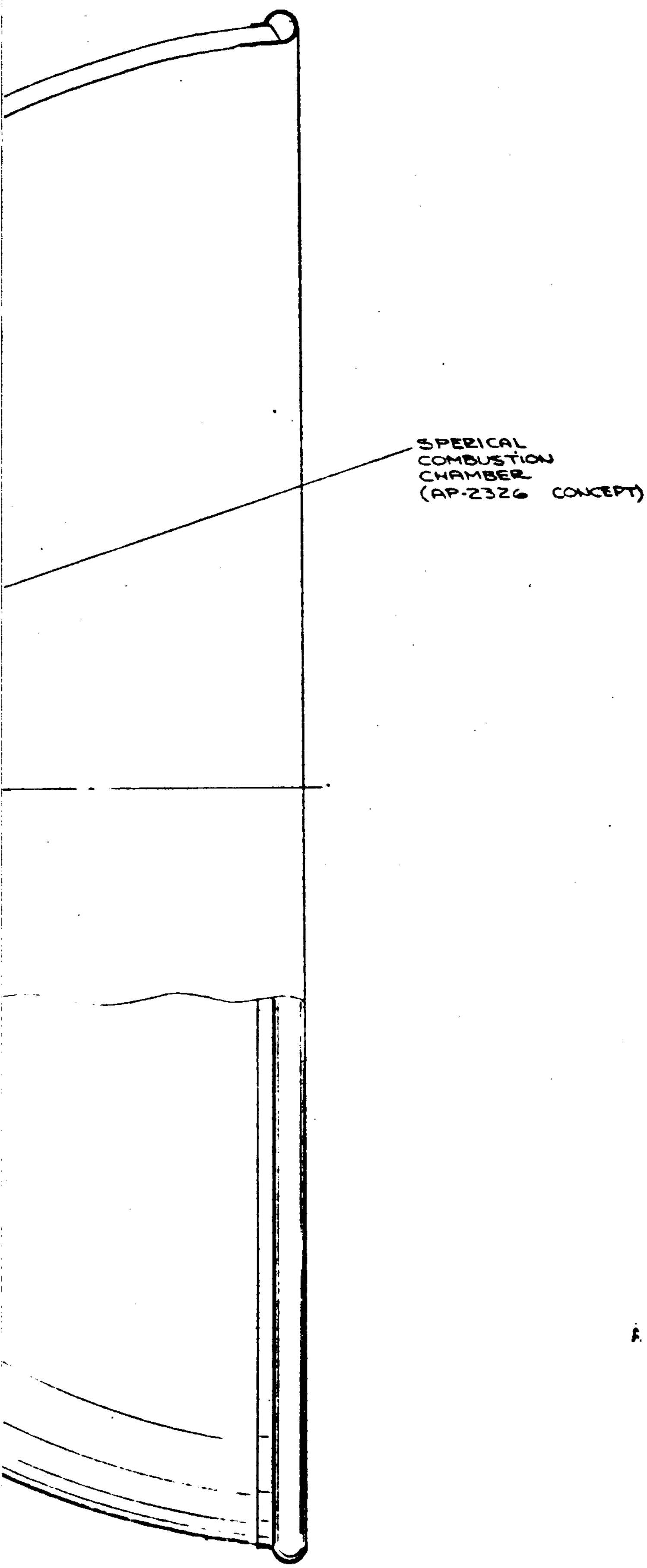


Figure 85

ADVANCED DESIGN

ROCKETDYNE

THRUST VECTOR CONTROL CONCEPT :- FIXED NOZZLE SKIRT WITH GIMBALED C.C.

HOCHE JR

SCALE NONE DATE 4-1-64

AP-2327

FOLD-OUT #2

Tap-off Methods. A brief investigation of possible methods of extracting turbine-drive gases from large toroidal combustors was made. The intent was to determine if other concepts could provide a greater measure of integration than the tap-off and plate concept (see discussion of O_2/H_2 booster system). It was assumed that these large (approximately 6M pounds thrust) toroidal combustors would be segmented.

There are two methods of tapping-off through the chamber wall. The method shown in Figure 86(a) consists of locally reducing the cross section of the chamber tubes to form holes through which the hot gas can pass. A collector manifold is placed on the outside of the tubes to direct the flow to the turbines. The main injector is designed to give cool gas in the tap-off region. This arrangement takes advantage of the cool region which is normally maintained near the tube wall. The chamber weight is increased only by the amount added in the collector manifold. During development, it will be necessary to move the tap-off location several times to determine the proper position. To change the location, new chamber tubes are necessary and an entire chamber and injector assembly must be made. Attaching the tap-off manifold to the tube wall may also be a problem.

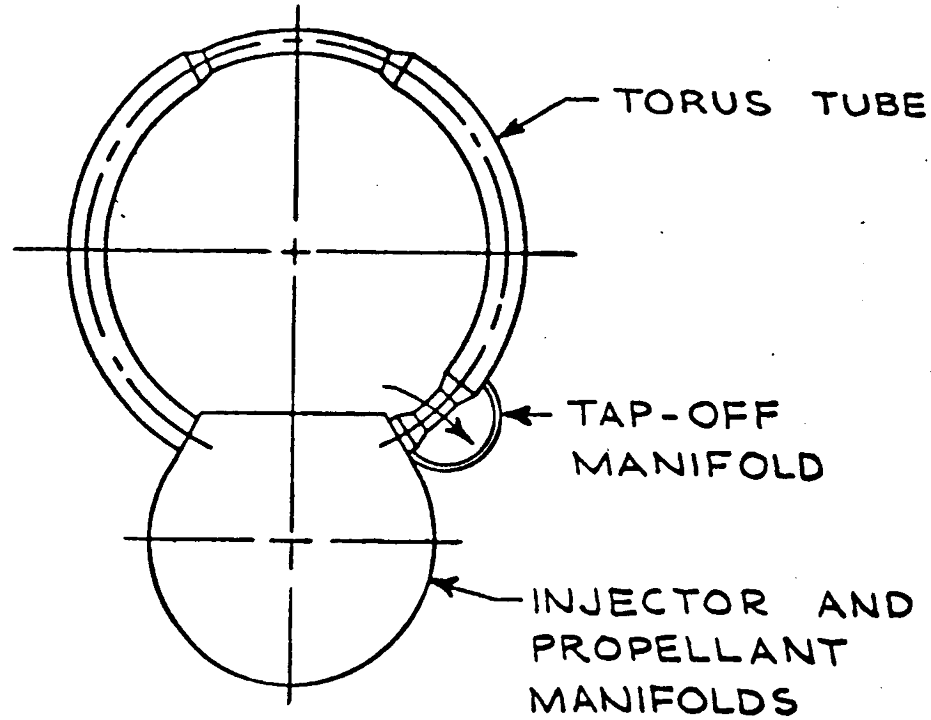
The method shown in Figure 86(b) is the same as Figure 86(a) except that a solid wall is used in the tap-off area. The main feature of the solid wall is easy relocation of the tap-off holes. Old holes could probably be welded up and new ones drilled. The temperature is low enough in the area of the solid wall to make regenerative cooling unnecessary. The solid wall provides a suitable attach point for the tap-off manifold. It is possible, however, that in some applications the tap-off manifold will be as large in diameter as the main chamber. This could easily cause packaging problems. The solid wall chamber should weigh slightly more than the tube wall chamber, Fig. 86(a).

Figures 87(a) and 87(b) show two variations of tapping-off through the injector. These schemes have the advantage of not affecting the chamber tubes. Figure 87(a) has the tap-off opening flush with the injector face. This method is probably unsuitable, because some unburned propellant could be taken from the chamber, and produce freezing or clogging in the tap-off lines.

In Figure 87(b), the tap-off location is raised from the injector face so cold propellants do not enter the tube. The length and location of the tube will have to be determined experimentally.

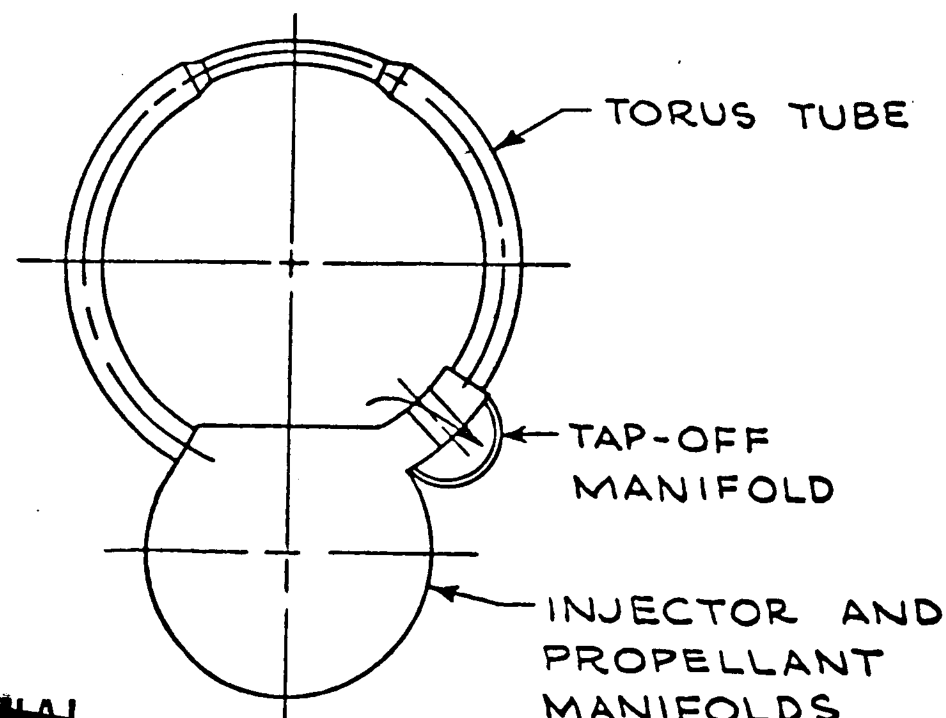
Passing hot gas through the injector may have an undesirable effect on the propellants. There should be no problems in attaching the manifold, but packaging of a large manifold may still be a problem. The modification of local mixture ratio can be achieved as described for system 1. The weight of these systems will be very close to that of system 1.

[redacted]



TUBE WALL

(a)



[redacted]

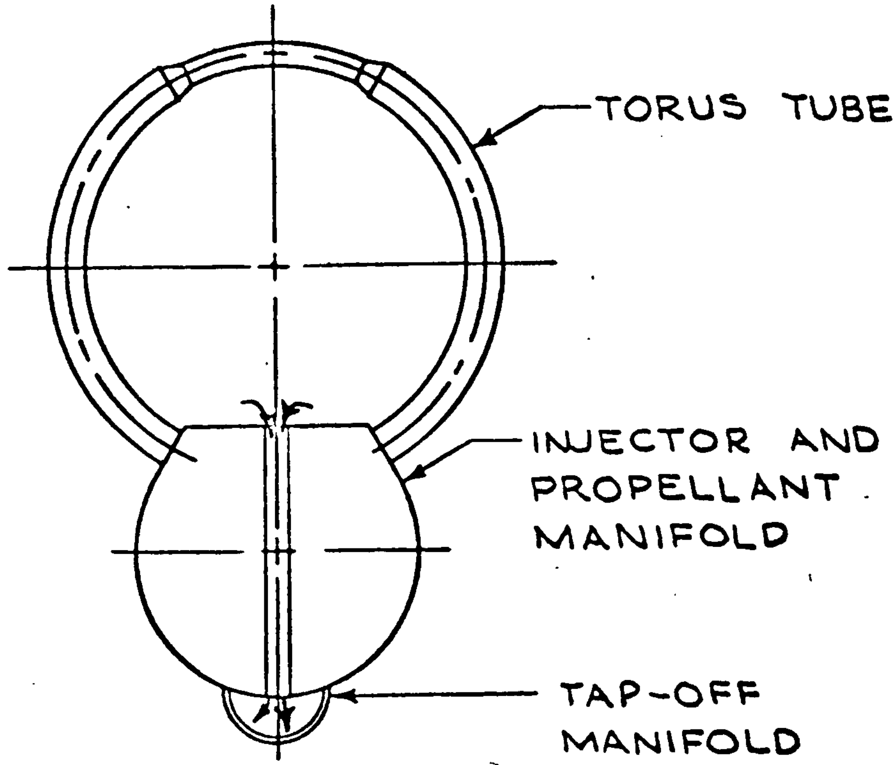
SOLID WALL

(b)

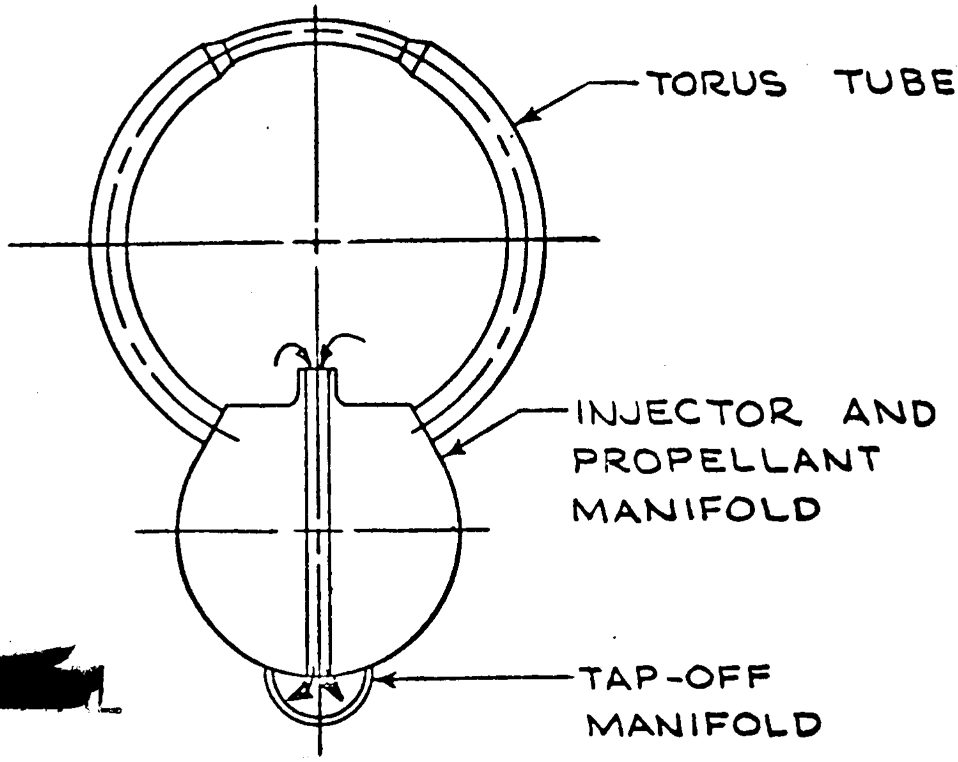
TAP-OFF METHODS

Figure 86

[redacted]



FLUSH INJECTOR FACE
(a)



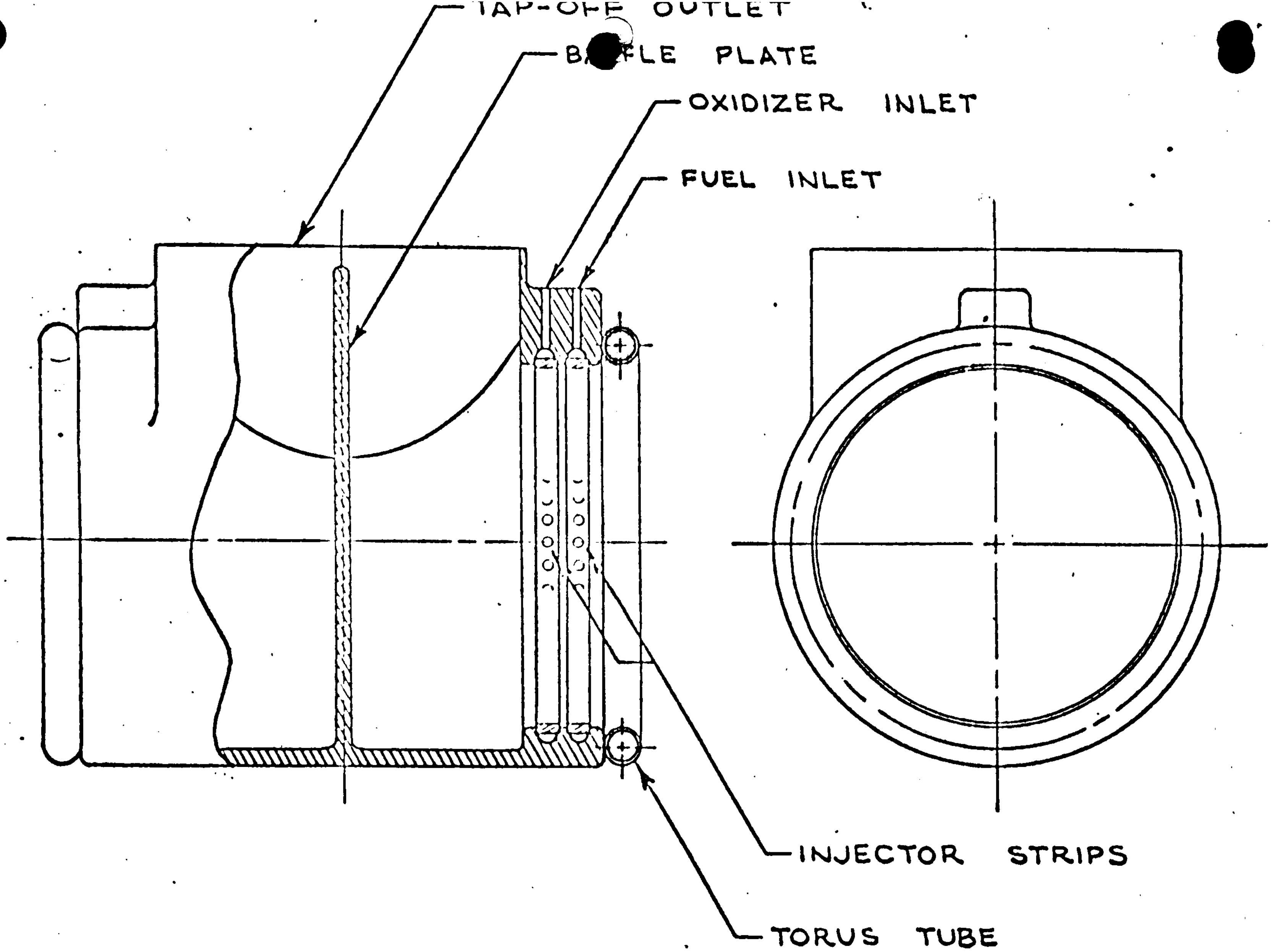
[redacted]

RAISED INJECTOR FACE
(b)

TAP-OFF METHODS
Figure 87

System 5 is the tap-off end plate (Fig. 88). The end plate is utilized as a location for removing tap-off gas. Since all of the gas is removed through one large hole, no collector manifold is necessary. Several variations of the tap-off end plate are possible. As shown in Figure 88 , the end plate has a baffle, a fuel injector and an oxidizer injector. With this configuration, the total tap-off flow can be provided by the end plate injectors. It is also possible to eliminate the oxidizer injector and inject just enough fuel to cool off the tap-off gas. The last method would be to produce a fuel rich area by modifying the main injector pattern. The baffle plate may be eliminated if desired, or one chamber segment can be completely isolated from the other. There should be very little weight addition with this system, because there is no collector manifold and the tap-off end plate should be about the same weight as the usual (assuming a segmented combustor is used) cooled end plate. Since there is no collector manifold, packaging problems should be minimized.

Based on the above discussion, it is concluded that the tap-off end plate concept is the most easily integrated of the concepts discussed and is probably the lightest. It should be noted that the above conclusion is not based on an exhaustive investigation, and must, therefore, be considered tentative pending a more detailed investigation of tap-off methods.



TAP-OFF END PLATE

Figure 88

ACKNOWLEDGEMENT

The contributions of the numerous technical personnel at Rocketdyne that have participated in this program, both in advisory and technical capacities, are gratefully acknowledged. Among those providing consultation and advice were: S. F. Iacobellis, S. R. Logan, A. J. Sobin and D. Mon[illegible]co. Direct technical contributions were made by: D. Bachman, R. Bremner, T. Cowell, T. Dea, H. Hoche, J. Huntsinger, H. Maeda, D. McGowan, R. [illegible]ooley, L. Russell, W. Shuker, and E. Sowell.

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